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SOCIETY

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The Journal of the American Rocket Society is devoted to the advancement of the field of jet propulsion through the publication of original papers disclosing new knowledge and new developments. The term "jet propulsion" as used herein is understood to embrace all engines that develop thrust by rearward discharge of a jet through a nozzle or duct, and thus includes systems utilizing atmospheric air and underwater systems, as well as rocket engines. The Journal is open to contributions, either fundamental or applied, dealing with specialized aspects of jet and rocket propulsion, such as fuels and propellants, combustion, heat transfer, high temperature materials, mechanical design analyses, flight mechanics of jet-propelled vehicles, astronautics, and so forth. The Journal endeavors, also, to keep its subscribers informed of the affairs of the Society and of outstanding events in the rocket and jet propulsion field.

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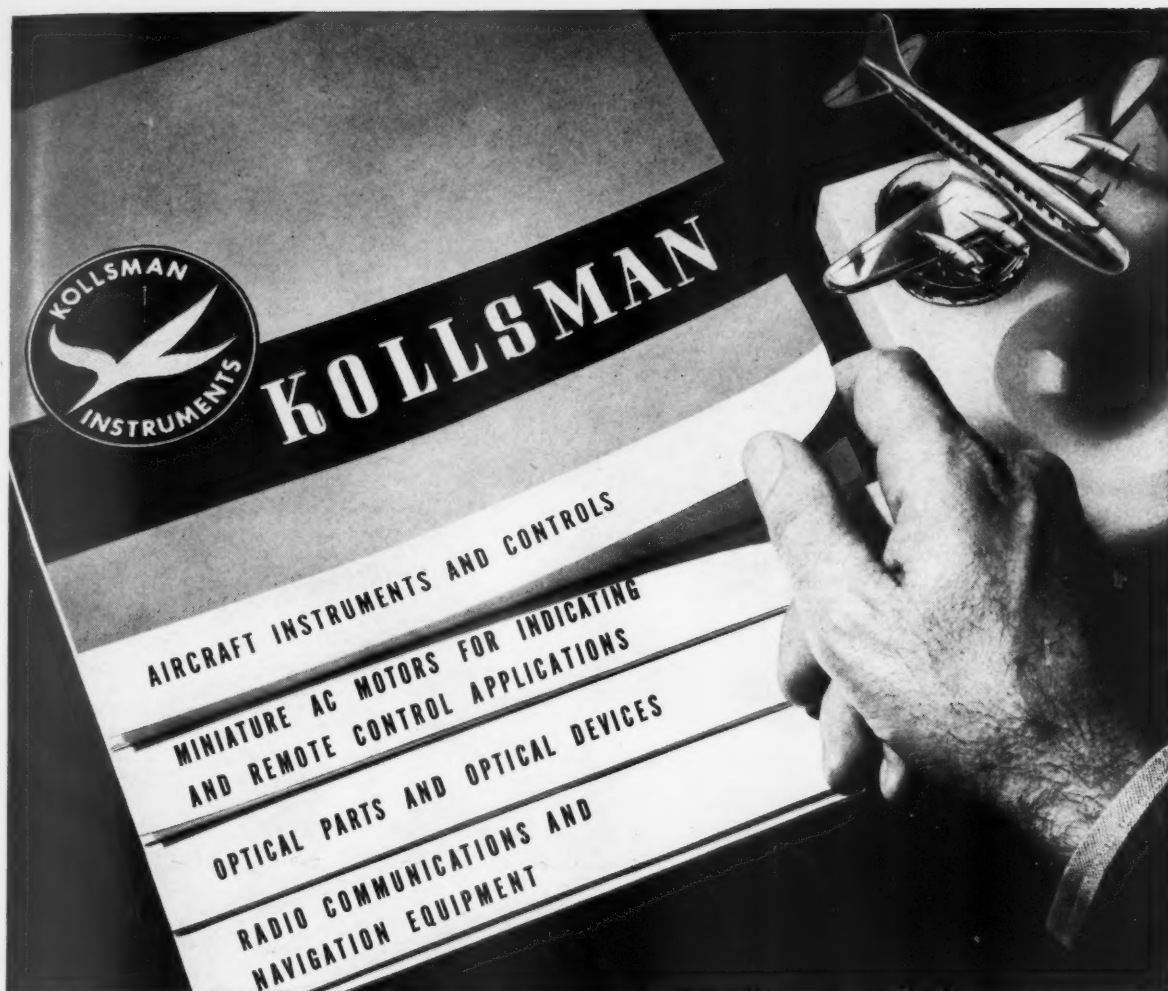
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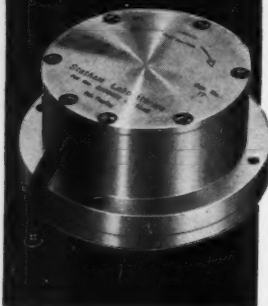
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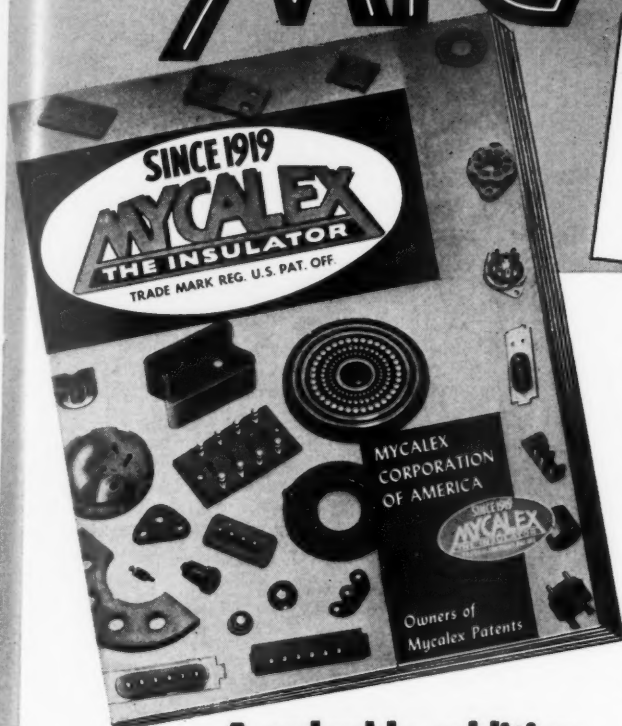
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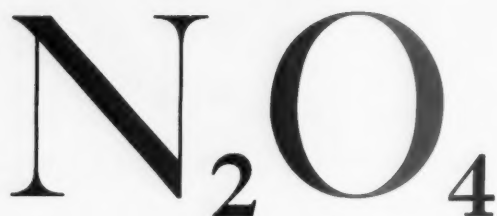
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# Large-Scale Production and Handling of Liquid Hydrogen<sup>1</sup>

H. L. COPLEN<sup>2</sup>

Aerojet Engineering Corporation, Azusa, Calif.

The design and operation of the first hydrogen liquefaction plant specifically constructed for the production of liquid hydrogen as a rocket propellant are described. This plant has a liquefaction capacity of over 12 lb per hr and is designed for continuous operation. A functional and physical description of the plant is given to emphasize important parameters.

The general design of a proposed large scale hydrogen liquefaction plant is also described. This plant is based on a closed helium refrigeration cycle having a liquefaction capacity of about 120 lb per hour.

The design of storage vessels suitable for handling 100-lb and 250-lb lots of liquid hydrogen is discussed, and basic design criteria and data are provided for use. Handling and safety precautions appropriate to liquid hydrogen are discussed, based on extensive experience gained as a result of the production and handling of approximately 7400 lb of liquid hydrogen during a period of about six months.

## I Introduction

### A HISTORICAL INTEREST

MANY of the early pioneers in rocket development studied the use of liquid hydrogen as a fuel, but the severe handling problems associated with its extreme physical properties were somewhat discouraging. Illustrative of these properties which make practical handling difficult are a boiling point of  $-426^\circ\text{F}$  at one atmosphere and a density about one seventh that of water. Interest in the methods and apparatus used in hydrogen gas liquefaction has recently become much more widespread because of a realization that handling methods may be developed to cope with the problems. In addition to the steadily increasing requirements for liquid hydrogen in the basic research field, new uses peculiar to the armed forces and to the Atomic Energy Commission have been discovered. In general, these new applications appear to require the use of advanced concepts in plant design in order to reduce plant volume and weight and to provide for reasonable portability.

In addition, consideration must be given to the potential need for very large scale plants. Most liquefaction cycles and plant component designs currently in use are based on commercial and necessarily conservative engineering practices and have not utilized recent advanced machinery and cycle design concepts for economic reasons.

The armed forces have potential requirements for both field and base generators of liquid hydrogen for supplying rocket-propelled missiles. The development of field generators would seem to be especially urgent because of the necessity for minimum equipment volume and low weight per unit output. Refrigeration machines for reducing storage losses from liquefied gas storage vessels may also be important, and work is now being done in this field.

The Atomic Energy Commission has an apparent potential need for gas liquefaction plants for low-temperature, solid-state research, for nuclear propulsion research, and for possible use in the production of isotopes of hydrogen. Any of these potential uses might require scales of operation never heretofore contemplated, particularly in respect to the liquefaction of hydrogen and helium gases.

### B DEVELOPMENT AT AEROJET

In the course of a rocket research and development program carried on in 1948 and 1949 at Aerojet, it was necessary to design, build, and operate a hydrogen liquefaction plant with a capacity of over 12 lb per hr. This plant is the largest known hydrogen liquefaction plant and the only one designed for continuous 24-hour operation. Extensive experience in the handling of low-temperature fluids was gained as a result of the production and handling of approximately 7400 pounds of liquid hydrogen during a period of about 6 months. This work will be described in some detail below.

As a part of the major purpose for which the plant to be described was built, a high-performance injector for use with the liquid-hydrogen liquid-oxygen propellant combination was developed on a 400-lb thrust scale. A 3000-lb thrust chamber was then developed which

<sup>1</sup> Presented at the American Rocket Society session at the 20th Annual Meeting of the Institute of the Aeronautical Sciences in New York, N. Y., February 1, 1952.

<sup>2</sup> Senior Engineer, Liquid Engine Division. Member ARS.



FIG. 1 3000-LB LIQUID HYDROGEN AND LIQUID OXYGEN MOTOR IN OPERATION

operated at specific impulse values ranging from 93 to 99 per cent of theoretical. The 3000-lb thrust chamber is shown in operation in Fig. 1.

The first pump to successfully produce high heads in pumping liquid hydrogen was built and tested at Aerojet during this same program. This pump is shown in Fig. 2. Liquid hydrogen pump tests were conducted

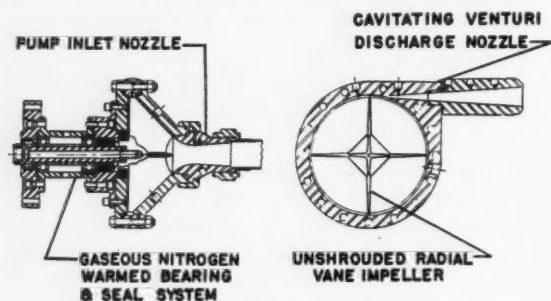


FIG. 2 EXPERIMENTAL LIQUID HYDROGEN PUMP

at shaft speeds to 35,000 rpm, producing heads to 15,000 feet (450 psi) at flow rates to 0.75 lb/sec. It was therefore demonstrated that pumping liquid hydrogen in a turbo-rocket engine is perfectly feasible and can be accomplished with a single-stage centrifugal pump.

## II Plant Design

### A AEROJET LIQUEFACTION PLANT

#### 1 General Design Criteria

This plant is designed to process and liquefy electrolytic hydrogen (and any isotopes that may be present) on a continuous-operation basis. Dr. H. L. Johnston of the Ohio State University Research Foundation acted as consultant in the plant design. A high-pressure, liquid-nitrogen-precooled, Joule-Thomson cycle is used. Dr. Johnston's cryostat design was adopted because it was close to the required capacity and because it was immediately available. Dual purification units are used in all cases where periodic regeneration is required. Safety and dependability in operation are provided to a marked degree by the ar-

rangement of equipment and controls and by the use of monitoring instrumentation to maintain gas purity and to protect against atmospheric contamination.

Two hydrogen compressors are used, each having a capacity of 94 cfm. These units may be used alternately or simultaneously to produce approximately double the single-compressor output of liquid hydrogen. The plant capacity with one compressor is 42 liters (6.6 lb) per hr. Liquid nitrogen is required at the rate of about 1.4 liters per liter of liquid hydrogen produced; about 1.0 liter is used in precooling and 0.4 liter for purifiers and other miscellaneous uses. This nitrogen usage may also be stated as about 34 equivalent cubic feet (at standard conditions) per liter of liquid hydrogen or about 1410 equivalent cu ft per hr.

The plant described below contains some equipment which represents a compromise required to permit the initial production of liquid hydrogen on schedule. The cycle used was necessarily selected for minimum installed cost because of the limited period of operation expected. Accordingly, the plant might be considered to be in a laboratory-pilot stage of development.

#### 2 Functional Description (see Fig. 3 for Plant Flow Diagram)

(a) Hydrogen gas is fed from the hydrogen tube banks  $A_1$  or  $A_2$  (capital letters in this report refer to the symbols in Fig. 3, unless otherwise indicated) or from the hydrogen tube trailer ( $A_3$ ) through the Deoxo catalyst purifier (B), to remove any oxygen that may be present. It enters the low-pressure system to replace the gas that has been liquefied in the process. On entry, it mixes with the gas returning from the liquefaction step and is compressed to approximately 2000 psig by either of two compressors ( $H_1$  or  $H_2$ ).

(b) The gas is then passed through a dryer set, consisting of a heat-exchanger condensation dryer ( $O_1$  or  $O_2$ ) and two silica gel adsorption dryers ( $L_1$  and  $L_2$ ,  $L_3$  and  $L_4$ ). The silica gel dryers are operated in series, one upstream and one downstream of a refrigeration evaporator heat exchanger (J). The flow of the hydrogen gas through the dryer set is as follows: (1) After passing through the condensation dryer and one of the silica gel dryers, it enters the refrigeration evaporator heat exchanger (J), where it is cooled to approximately  $-60^\circ\text{C}$  by means of Freon 22 refrigerant. (2) It then enters the second silica-gel dryer and, subsequently, the coil in the condensation dryer, where it cools the warm gas entering the shell side of this unit, thereby regaining most of the heat lost during the cooling and condensation process.

(c) The gas next enters the liquefier cryostat ( $V_1$  and  $V_2$ ), where a small portion is by-passed through the upper coil section of the nitrogen precooler ( $V_1$ ) while the remaining gas goes through the upper coil section of the hydrogen interchanger ( $V_2$ ). This procedure is followed in order to use the cooling effect of the return hydrogen gas most efficiently.

(d) The two incoming gas streams then join each other in the nitrogen precooler ( $V_1$ ), and proceed through the next cooler coil section and then through a



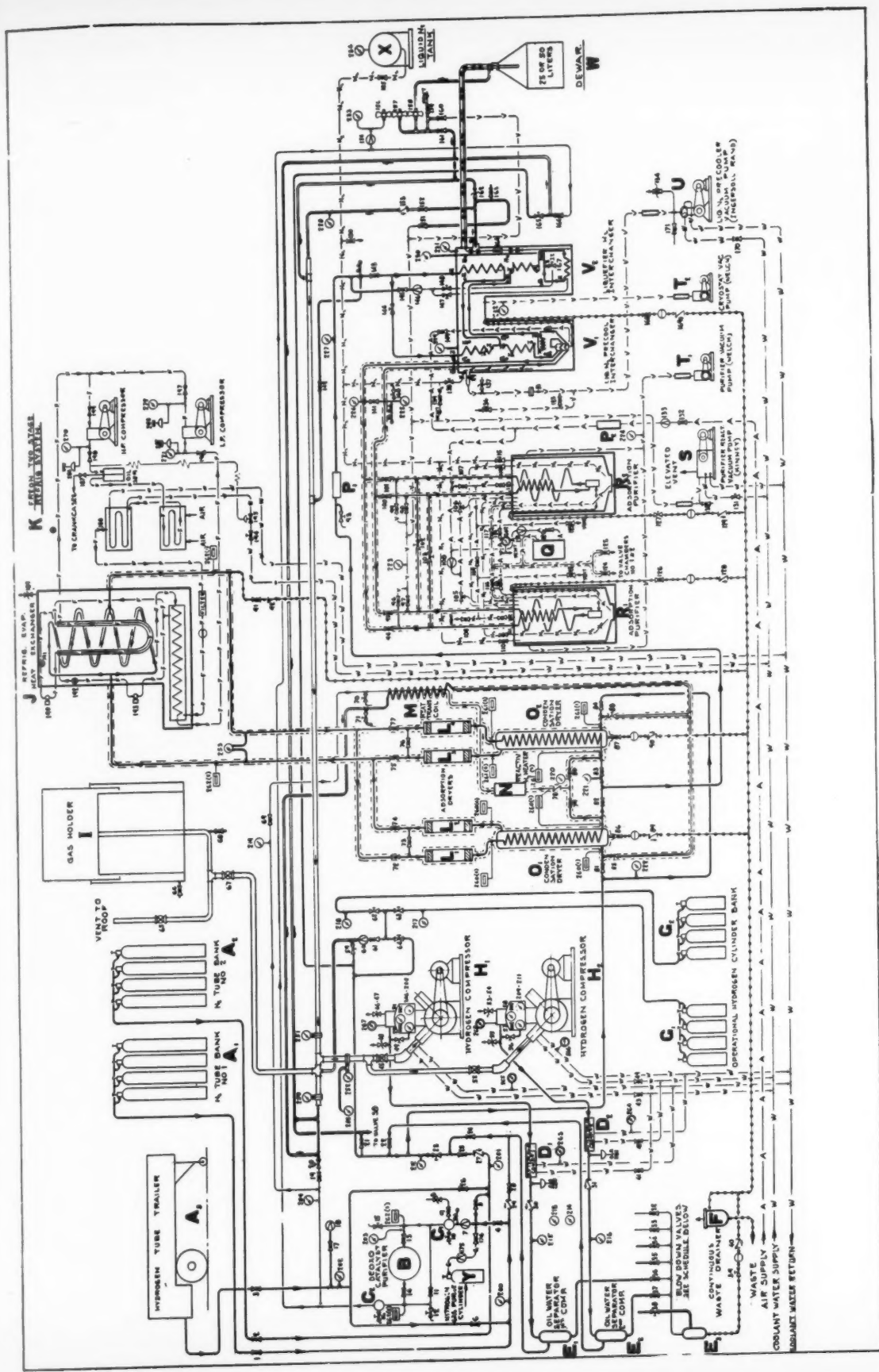


FIG. 3 HYDROGEN LIQUEFIER FLOW DIAGRAM

coil located in the bottom section of this unit, the latter being submerged in liquid nitrogen. The hydrogen then enters a liquid-nitrogen-cooled silica-gel adsorption purifier ( $R_1$  or  $R_2$ ) for additional purification by the adsorption of any oxygen or nitrogen impurities in the hydrogen gas at the liquid nitrogen temperature.

(e) The gas then re-enters the nitrogen pre-cooler ( $V_1$ ), where it passes through two coil sections submerged in liquid nitrogen, and then enters the hydrogen interchanger ( $V_2$ ) once more. In this unit it passes through the central coil section surrounded by effluent cold hydrogen gas and the lower coil which is submerged in liquid hydrogen.

(f) Finally, the hydrogen gas passes through a Joule-Thomson expansion valve, where approximately one fourth of the gas is liquefied, and the remaining gas re-enters the process cycle at low pressure after passing upward around the tubes of the heat-exchanger coil sections. The liquid hydrogen collects within the glass Dewar of the liquefier ( $V_2$ ) from which it may be transferred periodically or continuously as desired.

### 3 Physical Description

#### (a) Plant Installation

(1) The plant installation is shown in Fig. 4. The



FIG. 4 LIQUID HYDROGEN PLANT INSTALLATION

engine-driven compressors used in initial operations are seen on the left and the liquid nitrogen storage sphere on the right. All other plant components except for the gaseous hydrogen storage banks were within the sheet metal plant building.

#### (b) Hydrogen Storage Provisions

(1) *Portable Hydrogen Tube Trailer:* Gaseous hydrogen may be supplied to the plant using portable hydrogen tube trailers. Connections are provided so that gas can be transferred from the tube trailer to either of the two storage banks or directly to the liquefier system.

(2) *Hydrogen Storage Banks:* Two storage banks, each consisting of 36 cylinders and holding approximately 2600 cu ft of hydrogen (measured at 70 F and one atmosphere) at a pressure of 2000 psig, are used as the main storage for process gas. The cylinders in each bank are manifolded together into a common stainless steel line which serves as the feed line to the equipment located inside the liquefier building. Suitable valves and connections are provided whereby it is possible to transfer the gas from either bank into the hydrogen system.

(3) *Low-Pressure Gas Holder:* A floating gas holder (I), with a capacity of 200 cu ft, is installed in the liquefier building to maintain a constant pressure of about  $1\frac{1}{4}$  psig to the inlet side of the compressor. The holder acts as a surge tank for the cycle and is maintained at nearly mid-position by balancing the consumption and supply of the system. Tricresyl phosphate is used as the liquid sealant in the gas holder. This compound was selected for use because it has a low vapor pressure over the temperatures encountered, does not absorb an appreciable amount of oxygen from the air, and is non-combustible.

(4) *Service Storage Banks:* Two banks of four cylinders each are arranged to receive pure hydrogen gas from the system during shutdown of the liquefying process. This arrangement provides a supply of purified hydrogen which facilitates operations when the unit is started up after shutdown. Each of the cylinders has a capacity of about 215 cu ft (measured at 70 F and one atmosphere) at a pressure of 2000 psig.

#### (c) Hydrogen Compressors

A Chicago pneumatic compressor (three-stage, 94 cfm) supplies hydrogen to the system. One of the two units installed is shown in Fig. 5. An identical standby

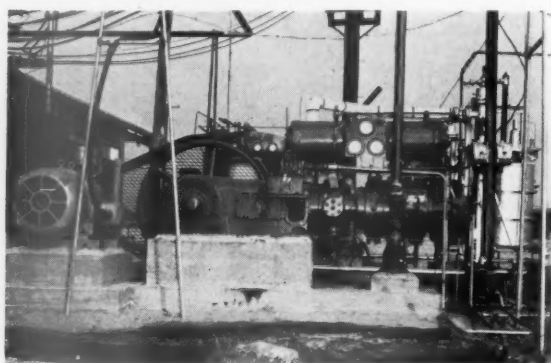


FIG. 5 THREE-STAGE 94-CFM HYDROGEN COMPRESSORS

compressor is connected in parallel. Both are driven by 60-hp electric motors, and the units may be simultaneously operated to nearly double the output capacity. Mercoid pressure switches, set at 2500 psi, protect the compressors from overload. Pressure-relief valves are located at each compressor stage. The compressors are automatically controlled by the volume of gas in the gas holder utilizing dual limit switches. The compressor will shut off when the gas holder is empty and will start when it is completely full. This system provides for disposing of any gas that boils off from the liquid hydrogen receiver (W) during plant shutdown or periods of intermittent operation. The gas is pumped from the gas holder back through a check valve into the supply banks.

#### (d) Oil and Water Separator

Two high-pressure oil and water separators are installed in the discharge lines of both hydrogen compressors ( $H_1$  and  $H_2$ ) to remove any oil or water droplets

which may be entrained in the hydrogen gas after compression.

#### (e) Purifier Equipment

(1) *Deoxo Catalyst Purifier*: This unit is used to remove oxygen from the raw hydrogen supply by means of a palladium catalyst. A pressure-regulating valve at the unit maintains its inlet pressure at 50 psig. This unit is a Baker Deoxo purifier with a capacity of 2500 cu ft per hr at 50 psig operating pressure. It is designed to reduce the oxygen content of the raw hydrogen gas to below one part in a million.

(2) *Dryer System*: Moisture is removed from the gas stream by means of two dryer sets. A complete dryer set consists of one condensation dryer and two silica-gel adsorption dryers mounted in a steel case. Santocel is used in the space between the three dryers and the case for insulation purposes. The two dryer sets are connected in parallel, to make possible the regeneration of one set while the other is in service.

(a) *Condensation Dryer* (Fig. 6): The gas enters

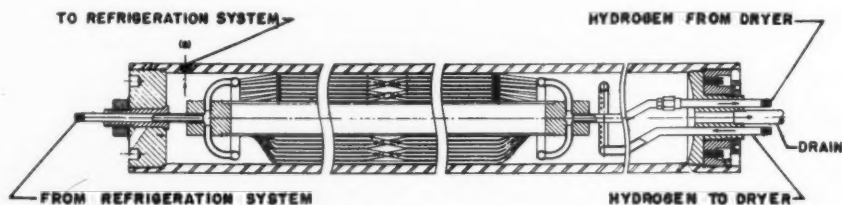


FIG. 6 HYDROGEN CONDENSATION DRYER

the dryer at the bottom through the shell inlet and passes through the shell around copper coils containing a return flow of cold gas that already has been dried in the same system and subsequently cooled in the Freon 22 refrigeration evaporator heat exchanger (J). Moisture in the gas is condensed on the outside of these cold coils and is removed from the system by means of drain valves. There are 19 parallel copper-tubing coils in the unit arranged according to the Giauque system.

(b) *Refrigeration Evaporator Heat Exchanger*: The temperature of the hydrogen emerging from the refrigeration evaporator heat exchanger is approximately  $-60^{\circ}\text{C}$ . This heat exchanger consists of 12 turns of 1-in. type-K copper tubing (7 turns in an upper interchanger and 5 turns in the lower interchanger). Eight  $\frac{3}{16}$ -in. copper tubes connected in parallel to carry the

high-pressure gas are soldered along the outside of the 1-in. tubing for its entire length. Hydrogen gas entering the unit separates into two streams, one passing through the upper interchanger and the other through the lower interchanger. These streams join again after leaving the unit. This heat exchanger is housed in a steel case, which is packed with Santocel for insulation. Also included in the steel case are a U-shaped Freon subcooler and a Freon heat exchanger.

(c) *Refrigeration System*: A two-stage Freon 22 system of refrigeration has been included in the liquefying process to cool the hydrogen gas, leaving the shell side of either condensation dryer, so that, in its return through the tube side of the same dryer, it will cause moisture to condense from the incoming, ambient hydrogen gas. The flow cycle used may be followed by referring to the refrigeration system shown in Fig. 3.

(d) *Adsorption Dryer* (Fig. 7): Silica gel-type dryers are used to adsorb impurities that are not removed in the condensation dryers. Two sets of these

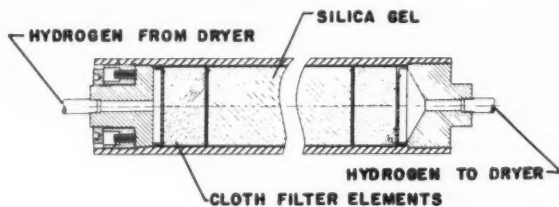


FIG. 7 SILICA ADSORPTION DRYER

adsorption dryers are employed. Each set consists of two dryers operated in series, one upstream and one downstream of the refrigeration evaporator heat exchanger (J).

(3) *Adsorption Purifier* (Fig. 8): Two purifiers, which may be connected in parallel or series, are utilized

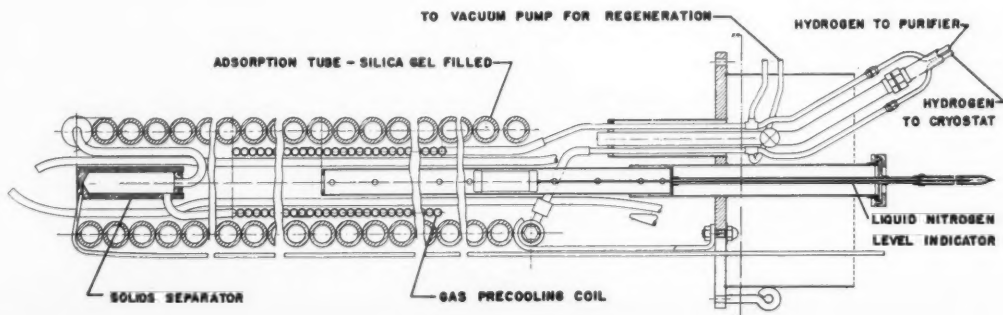


FIG. 8 ADSORPTION PURIFIER

to remove any foreign gases, primarily nitrogen, which may be present. Normally the purifiers are connected in parallel so that one of the units may be regenerated while the other is in the process cycle. Gas entering the adsorption purifier first passes through an inner coil of  $\frac{3}{8}$  in. copper tubing submerged in liquid nitrogen to cool the gas for maximum adsorption efficiency. It then enters a condensation separator after which the gas passes through an outer coil filled with silica gel where impurities are adsorbed. This outer coil is made of copper pipe ( $1\frac{1}{4}$  in. OD) with a total length of about 66 ft.

(f) **Liquefier Cryostat Assembly (Fig. 9)**

(1) The liquefier cryostat assembly consists of a liquid-nitrogen precooler interchanger ( $V_1$ ) and a liquid-hydrogen interchanger ( $V_2$ ) connected by a horizontal passageway that is evacuated during operation for the purpose of insulating the tube connections which pass between the two units. The coils of the units are contained in two separate 54-in. Dewars. The upper two coil sections of both units are wrapped with a thin Monel sheet, and a wool material is wrapped over the Monel sheets to provide a snug fit against the Dewars. The wrapping described above is provided to effect more efficient contact between the rising nitrogen or hydrogen gases and two upper coil sections of both units. At the upper part of the top coil sections of both units, a copper, funnel-shaped seal serves to prevent the effluent gases from by-passing the heat transfer surface in passing to the outlet connections. Thirty-one thermocouples are provided in the cryostat system. The coils and Dewars of both units may be removed from their containers for inspection and overhaul by means of a hydraulic lift.

(2) There are a total of five copper-coil interchangers in the liquid nitrogen precooler ( $V_1$ ). During normal operation, three of these are immersed in liquid nitrogen and the other two are cooled by effluent nitrogen gas. Copper manifolds are located in the inlet and outlet lines of the latter two unsubmerged coil sections. The liquid-nitrogen level may be observed through an unsilvered opening in the Dewar and it may also be determined by means of a level-indicator float provided in the unit.

(3) The liquid-hydrogen interchanger ( $V_2$ ) contains three copper coil interchangers. One is immersed in liquid hydrogen, and the remaining two are cooled by cold effluent hydrogen gas during normal operation. Two manifolds are provided for gas which enters and leaves the top coil interchanger. The liquid-hydrogen level may be observed through a similar unsilvered opening in the Dewar provided for the product liquid.

(g) **Vacuum Pumps**

(1) *Purifier Reactivation Vacuum Pump:* A Kinney, 27 cfm, rotary-type, water-jacketed high-vacuum pump is used primarily to remove all adsorbed gases from either of the adsorption purifiers during regeneration. In addition, this pump is used to evacuate the liquid hydrogen receiver and the shell side of the liquefier

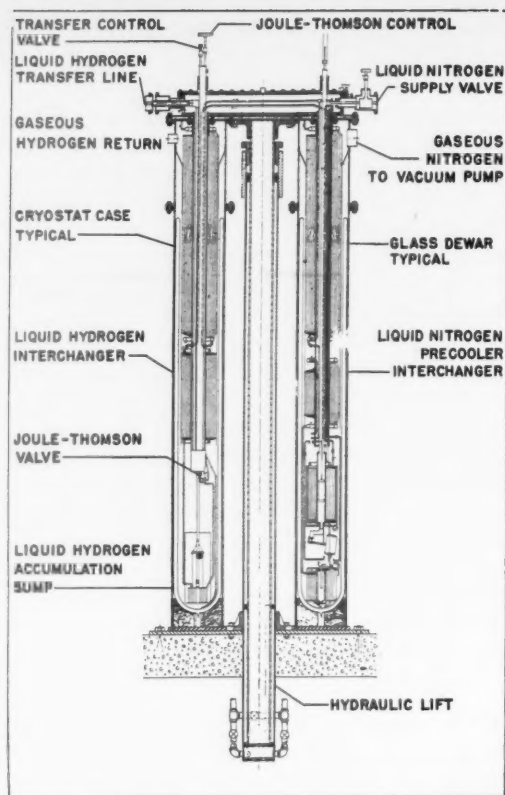


FIG. 9 GENERAL PLAN OF CRYOSTAT ASSEMBLY

hydrogen interchanger ( $V_2$ ) during flushing procedures.

(2) *Purifier Vacuum Pump:* A Welch, duo-seal, rotary-vane-type vacuum pump is used to evacuate the jackets of the adsorption purifiers ( $R_1$  and  $R_2$ ) simultaneously for insulation purposes.

(3) *Cryostat Vacuum Pump:* Another Welch, duo-seal, rotary-vane-type vacuum pump is used to evacuate and seal the horizontal passageway between the nitrogen and hydrogen sides ( $V_1$  and  $V_2$ , respectively) of the cryostat.

(4) *Liquid Nitrogen Precooler Interchanger Vacuum Pump:* An Ingersoll-Rand, 250 cfm, piston displacement, double-acting, single-stage vacuum pump is used to maintain a vacuum of about 25 in. of mercury in the liquid-nitrogen precooler ( $V_1$ ). This is done to lower the boiling point of the liquid nitrogen to about 64.7 K, which is just above its freezing point.

(h) **Heater Equipment**

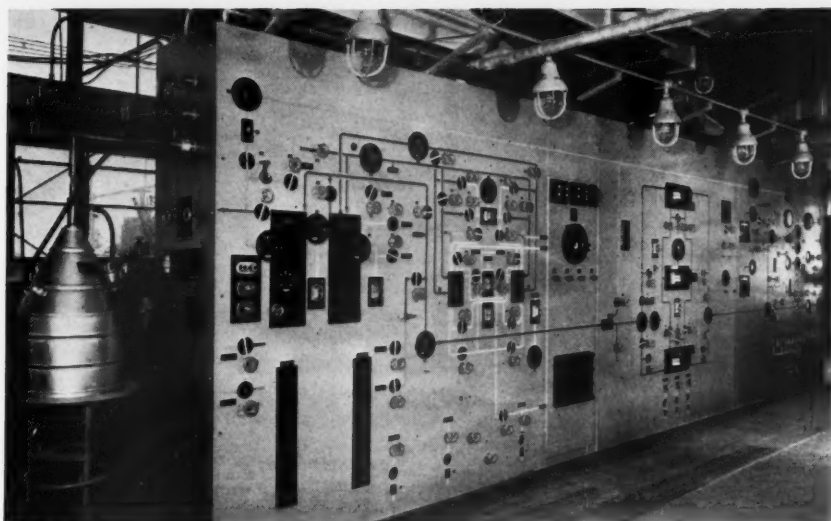
(1) *Dryer Reactivation Hydrogen Heater:* Removal of occasional plugs of frozen impurities in the lines and valves of the system can be accelerated by use of hot hydrogen gas. A 2-kilowatt, Chromolox, explosion-proof heater is provided from regeneration of the two dryer sets and to warm parts of the system for the purpose of removing frozen impurities. A temperature control is set to maintain an outlet hydrogen temperature of about 350 F, and a limit switch is set to cut the heater off at 400 F.

(2) *Purifier Reactivation Compressed Air Heater:* A similar Chromolox heater is used in the regeneration



FIG. 10

CENTRALIZED LIQUEFIER  
CONTROL PANEL



of the adsorption purifiers. Air at approximately 15 psig is heated to approximately 300 F. This heated air may also be used to remove plugs which may form in the valves or tubing of the purifier valve chambers.

(i) **Liquid Hydrogen Receivers**

Three 50-liter and three 25-liter vacuum Dewars are used as receivers for the liquid hydrogen during test operation. In addition, a container having no insulation is used for test operation; the liquid hydrogen is thus evaporated immediately upon its transfer from the liquifier side of the cryostat ( $V_2$ ). The design of large-capacity receivers for 100 and 250 lb of liquid hydrogen is described in the Section on storage and handling.

(j) **Liquid Nitrogen Supply and Storage**

The liquid nitrogen required for operation of the hydrogen liquefaction plant is stored in a 250-gallon Dewar which has a working pressure of 50 psig. A double-wall insulated line delivers the liquid to the various hydrogen liquefier units.

(k) **Control Panel (Fig. 10)**

The liquefying process is controlled primarily from a  $1\frac{1}{4}$ -in. steel panel, 8 feet high and 28 ft. long. On this panel is constructed a color-coded system flow diagram from which the use of each valve, gage, and unit can easily be understood. The panel not only supplies a systematic means of operating the liquefier equipment, but also provides a safety shield for the protection of operating personnel. Safety observation windows are provided through which both the liquid hydrogen and nitrogen levels in the cryostat may be seen.

(l) **Test Equipment**

(1) *Beckman Oxygen Analyzer:* The quantity of oxygen present as an impurity in the make-up hydrogen gas is continuously detected and recorded by a Beckman oxygen analyzer. The operation of the analyzer depends on the fact that oxygen is strongly paramagnetic, whereas the other common gases are, with few exceptions, slightly diamagnetic.

(2) *Alnor Dew Pointer:* (a) In the regeneration of the dryer sets, it is desirable to know when the moisture has been removed from the unit, so that it may be placed back in service when needed. To ascertain the completeness of removal, a dew-point analysis is made several times during the regenerative process by means of an Alnor Dew Pointer. The gas to be tested is fed to the instrument at a low pressure somewhat above atmospheric. The chamber is then rapidly exhausted to the atmosphere, with the resultant expansion and cooling of the gas. Several gas samples are taken and expanded from various initial pressures to determine the pressure at which fog begins to form. The expansion is adiabatic since it is accomplished almost instantaneously. This dew-point figure is then compared with the dew-point determined from a sample taken before the gas enters the dryer to determine its state of regeneration.

**4 Operation Record**

(a) The plant as originally designed had a production capacity of 4.7 lb (30 liters) per hr and was so operated from September 1948 to March 1949. Because of the increase in propellant requirements resulting from the engine specification change, the plant equipment was revised to increase the capacity to approximately 12 lb (76 liters) per hr, and it was operated on a three-shift basis at this capacity from March to June 1949.

(b) From September 1948 to June 1949 the gross plant production was 7400 lb (47,000 liters) of liquid hydrogen, with about 5300 lb (33,700 liters) being produced in the last four months of this period.

**B PROPOSED LARGER SCALE DESIGN**

For the purpose of a preliminary cycle analysis, a capacity of about 3000 lb per day was selected for use in the design. This scale was selected as being the probable lower capacity limit desirable for use in the development of a prototype plant on the basis of the size of the turbo-expander.

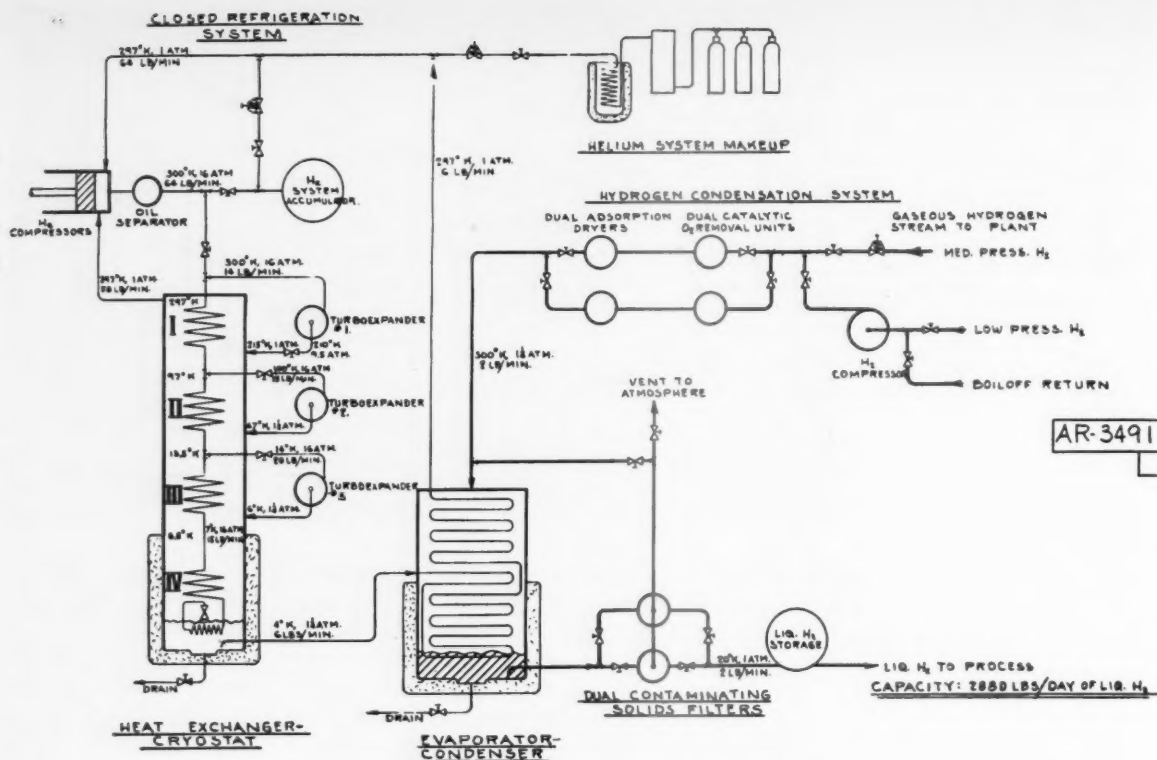


FIG. 11 PROPOSED CYCLE FOR LARGE SCALE LIQUEFIER

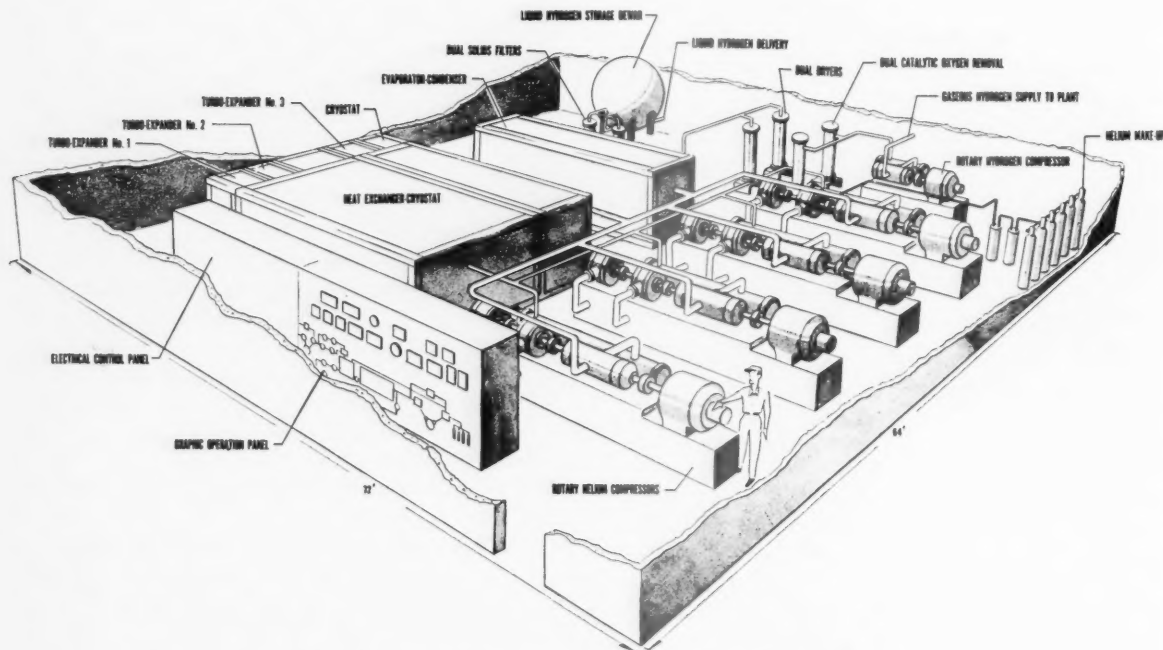


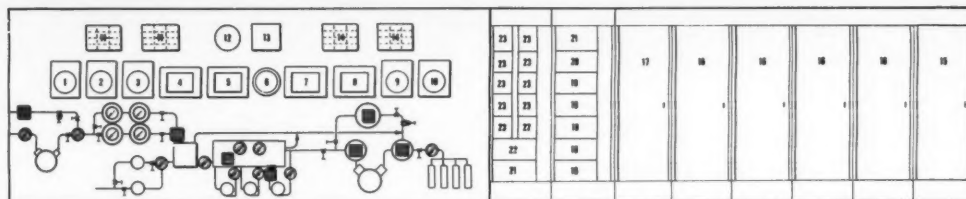
FIG. 12 EQUIPMENT LAYOUT—LARGE-SCALE HYDROGEN LIQUEFACTION PLANT

The proposed cycle for the large-capacity prototype plant is shown diagrammatically in Fig. 11. A closed refrigeration system operating at a maximum pressure of 16 atm (235 psia) is used to condense the gaseous hydrogen at nearly atmospheric pressure. Three turbo-

expanders are used at steadily decreasing temperature levels in the refrigeration system. Helium has been selected as the refrigerant on the basis of low required system pressures, inertness, and low initial cost. Most of these proposed plant features have been mentioned in

# GRAPHIC OPERATION PANEL

# ELECTRICAL CONTROL PANEL



## RECORDED SYSTEM DATA

- |  |                                 |   |
|--|---------------------------------|---|
| ① HYDROGEN FLOWS                         | ⑩ HELIUM SYSTEM TEMPERATURES    | ⑪ MAIN ELECTRICAL DISCONNECT                    |
| ② HYDROGEN DEWPOINT                      | ⑪ LIQUEFIED GAS LEVELS          | ⑫ REDUCED VOLTAGE STARTER, HELIUM COMPRESSORS   |
| ③ PARTS PER MILLION OXYGEN CONTAMINATION | ⑫ HELIUM FLOWS                  | ⑬ REDUCED VOLTAGE STARTER, HYDROGEN COMPRESSOR  |
| ④ RAW HYDROGEN ANALYSIS                  | ⑬ HYDROGEN SYSTEM ANNUNCIATORS  | ⑭ OVER-LOAD BREAKER, HELIUM COMPRESSORS         |
| ⑤ HYDROGEN SYSTEM TEMPERATURES           | ⑭ CLOCK                         | ⑮ OVER-LOAD BREAKER, HYDROGEN COMPRESSOR        |
| ⑥ COMBUSTIBLE GAS ALARM                  | ⑮ COMBUSTIBLE ALARM ANNUNCIATOR | ⑯ COMBINATION STARTER - CRYOSTAT VACUUM PUMP    |
| ⑦ TURBOEXPANDER TACHOMETERS              | ⑯ HELIUM SYSTEM ANNUNCIATORS    | ⑰ COMBINATION STARTER - SOLIDS FILTER HEATER    |
|  |                                 | ⑱ COMBINATION STARTER - SILICA GEL REGENERATION |
|  |                                 | ⑲ MISC. VACUUM PUMPS & LIGHTS                   |
- RECORDING PRESSURE GAGE
 INDICATING PRESSURE GAGE

FIG. 13 CONTROL PANELS FOR LARGE-SCALE HYDROGEN LIQUEFACTION PLANT

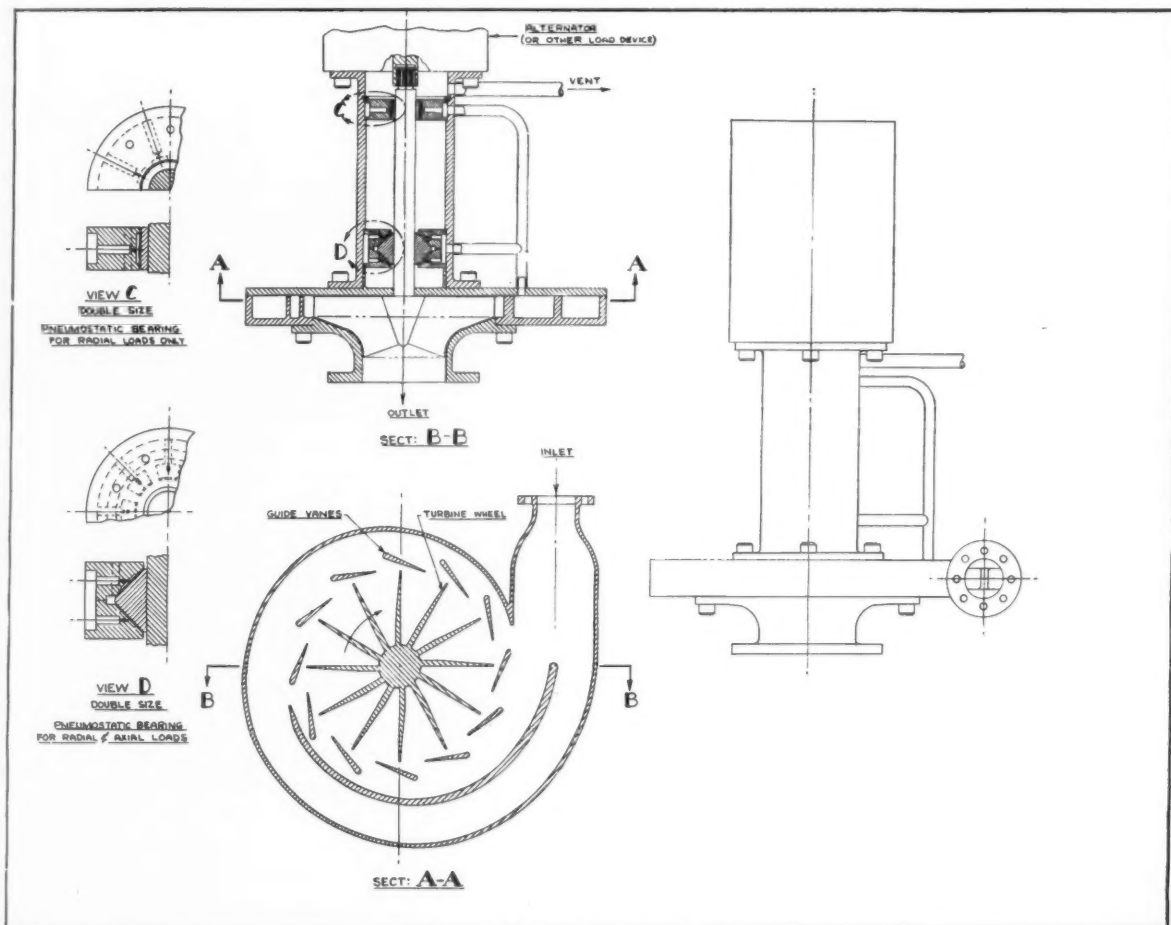


FIG. 14 HELIUM TURBO-EXPANDER

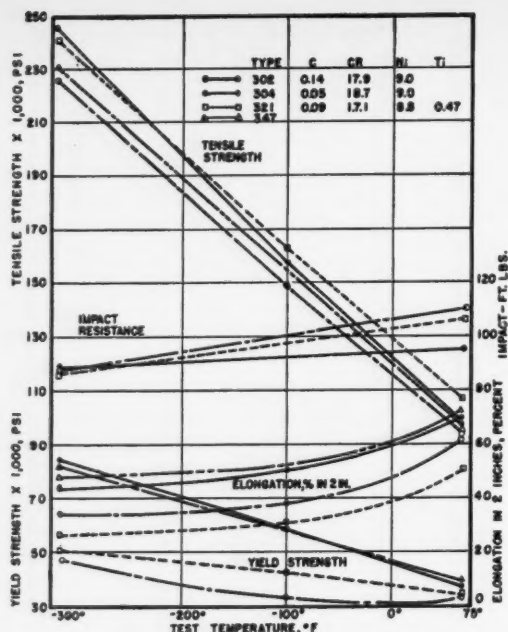


FIG. 15 TEMPERATURE EFFECT ON TENSILE PROPERTIES

the literature. References 1 to 5<sup>3</sup> are some sources of background on the use of work extraction for the cooling and liquefaction of gases.

In this cycle, the raw gaseous hydrogen is purified of oxygen content by dual catalytic means and any excessive water vapor content is removed by adsorption purifiers. Final hydrogen purification is accomplished by means of filters for the removal of all other contamination in the solid phase. Dual units are used for alternate regeneration.

A possible arrangement of the plant is shown in Fig. 12. Control panels for this plant are shown in Fig. 13. A graphic arrangement of the cycle controls is proposed to facilitate both operation and training. All important features of operation are recorded. For the 120-lb/hr (2880-lb/day) plant capacity, the power requirement will be approximately 1600 hp.

A preliminary design of a helium turbo-expander is shown in Fig. 14. Design operating speeds in excess of 60,000 rpm are believed practical with the "pneumo-static" bearing design illustrated. This type of bearing has proved satisfactory in principle by tests on "hydrostatic" designs now in use. The total plant turbo-expander horsepower is less than 10 per cent of the helium compressor horsepower so that this power probably will be dissipated by some convenient means such as a water brake or an electric generator.

The advantages of the proposed prototype cycle may be summarized as follows:

(1) No auxiliary precoolants or precoolant production apparatus are needed. (2) Dependability of basic system assured by the use of the closed-cycle refrigerant system. (Refrigerant gas purification necessary only on the makeup gas.) (3) High efficiency, lightweight

<sup>3</sup> References are listed at end of the paper, page 338.

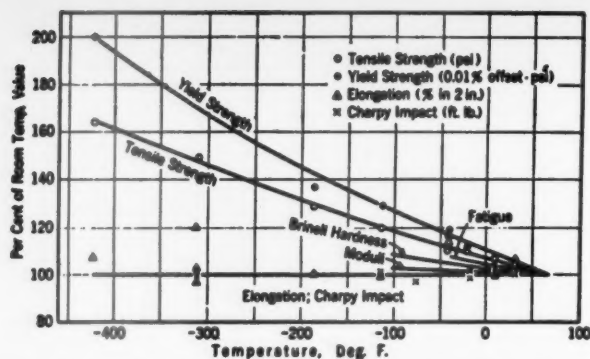


FIG. 16 PERCENTAGE CHANGE OF PHYSICAL PROPERTIES OF MONEL

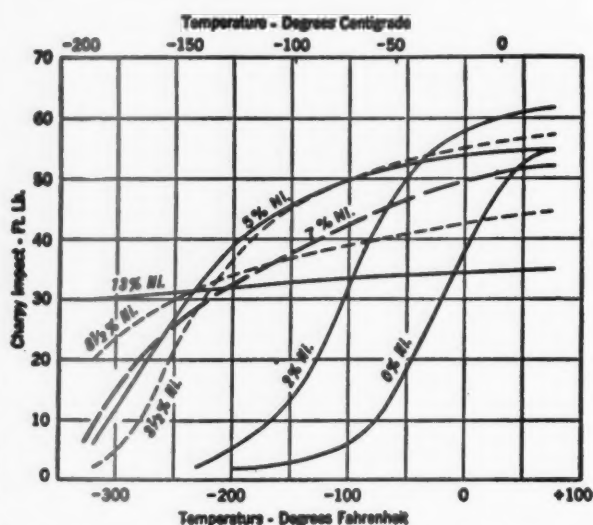


FIG. 17 EFFECT OF NICKEL CONTENT ON RESISTANCE TO LOW TEMPERATURE EMBRITTLEMENT

turbo-expanders used instead of expansion engines. (4) Liquid-solid phase separation used for purification of condensed gas. (5) Low system pressures: Maximum pressure helium in system, 235 psia; maximum pressure hydrogen in system, 20 psia. (6) Basic cycle is readily scalable to any desired production rate. (7) Basic cycle can be used to liquefy gas.

### III Storage and Handling

#### A LIQUID HYDROGEN STORAGE-VESSEL DESIGN

##### 1 Materials for Construction

Preferred materials for liquefied-gas storage vessels are the austenitic 18-8 stainless steels, Inconel, Monel, nickel, and low-carbon, high-nickel steels. Fig. 15 gives the temperature effect on tensile properties of the stainless materials. The effect of temperature on the properties of Monel are shown in Fig. 16. The effect of the nickel content on the low-temperature embrittlement of low-carbon steels is shown in Fig. 17.

##### 2 Insulation

(a) Work performed at Ohio State University on thermal conductivity through an evacuated space,



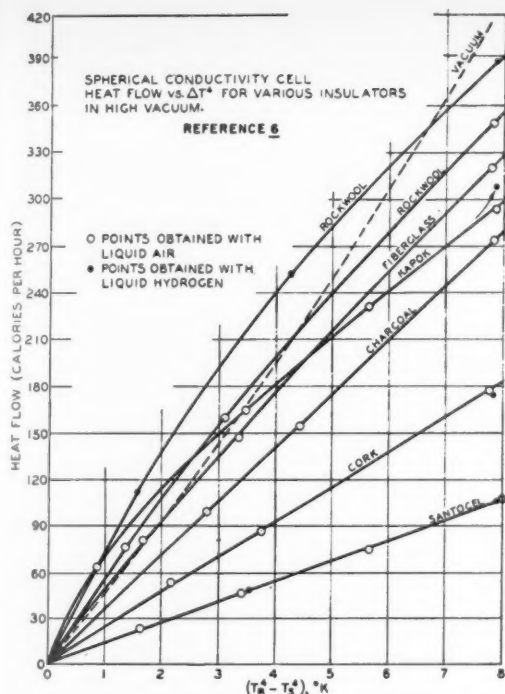


FIG. 18 COMPARISON OF VARIOUS INSULATORS IN HIGH VACUUM

with and without various insulating materials, has shown that the use of powdery insulators such as Santocel, cork, and charcoal is more efficient than the vacuum alone. Fig. 18 shows the effect of the introduction of various insulators on heat transfer through a highly evacuated space. Fig. 19 shows heat flow vs. pressure, and it is seen that the insulated space should be evacuated to 100 microns or below to retain reasonably good insulation.

(b) Introduction of a radiation shield in the form of a liquid-nitrogen jacket or a liquid-nitrogen cooled metal shield will reduce the heat flow through the jacket to about 1/30 of the unshielded value. For long-term storage at small-capacity liquefaction plants, the radiation shield is considered essential.

### 3 Specific Designs Utilized

(a) *High Pressure:* (1) A high-pressure storage vessel of 100-lb (25 cu ft) storage capacity was designed and built for dual use, both as a plant storage and accumulation vessel, and for use with pressures up to 1000 psi in motor and pump testing for engines up to 3000-lb thrust scale. The tank was arranged for forked-lift truck transport to the rocket test bays, so that the liquid-hydrogen transfer with its attendant losses would not be necessary. This tank is shown in Fig. 20. Two of these units were built for alternate services in the plant and test bays.

(2) The tank is composed of an inner pressure vessel having an 0.375-in. wall of heat-treated K Monel, surrounded by a Monel vacuum jacket filled with Santocel insulation. Surrounding this is a Monel vacuum-jacketed vessel which is filled with liquid nitrogen to precool the inner tank and reduce its thermal differential.

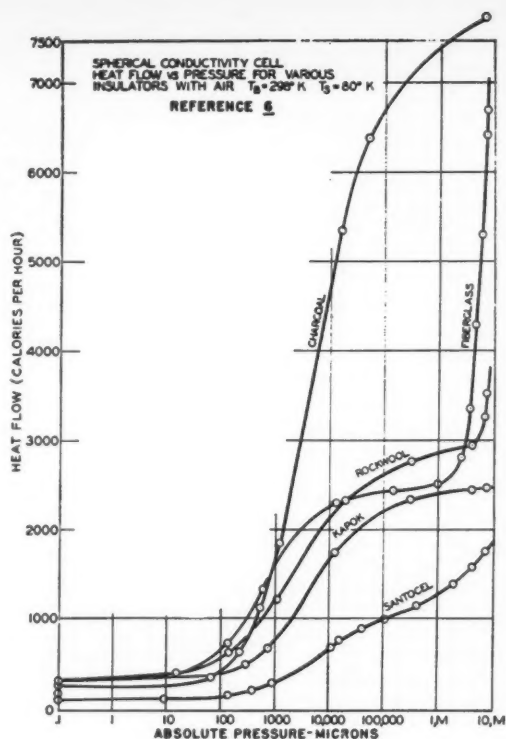


FIG. 19 IMPORTANCE OF DEGREE OF EVACUATION OF JACKET WITH VARIOUS INSULATORS

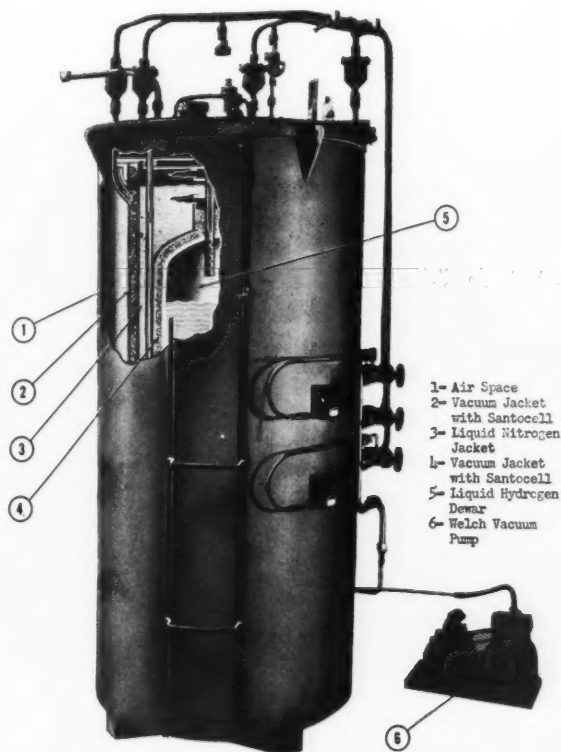


FIG. 20 HIGH-PRESSURE LIQUID HYDROGEN RECEIVER

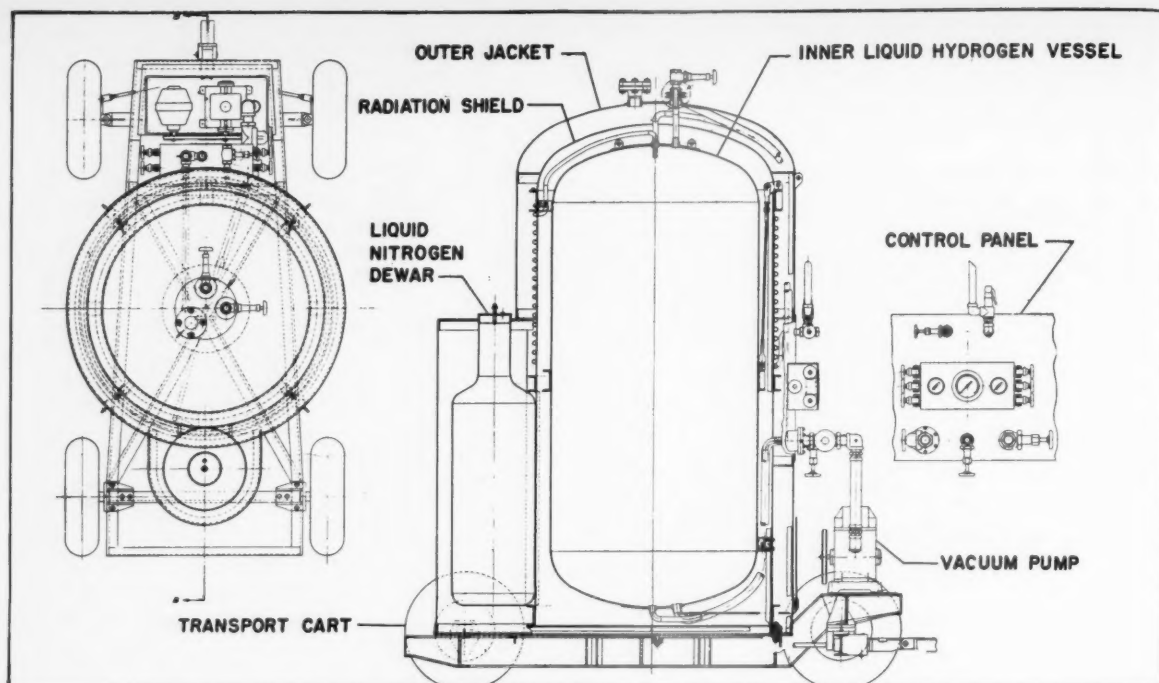


FIG. 21 MEDIUM-PRESSURE LIQUID HYDROGEN RECEIVER

(b) *Medium Pressure:* A medium pressure storage vessel of 250-lb (60 cu ft) storage capacity was designed for dual use, both as a storage and accumulation vessel and for use to 100 psi in pump and pumped rocket-engine testing. This unit (Fig. 21) is designed with a single vacuum jacket and a nitrogen-cooled radiation shield which is regeneratively cooled by the boil-off of hydrogen gas in storage.

(c) *Small Capacity:* Dewars for storage of small quantities of liquid hydrogen (the standard Purox and Hoffman Dewars are suitable) are made of all-metal construction and are available in sizes ranging from 15 to 100 liters.

#### 4 Storage-Loss Analyses

(a) Hydrogen exists in two recognized forms, ortho and para, which differ in the manner in which the spins of the atoms are coupled in forming the molecule. The para form of spin coupling is associated only with even

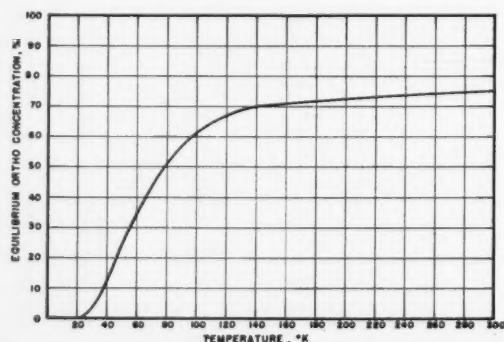


FIG. 22 EQUILIBRIUM ORTHO HYDROGEN AND CONCENTRATION

rotational energy states of hydrogen, and the ortho form only with odd rotational energy states. The equilibrium ortho-hydrogen concentration is plotted as a function of temperature in Fig. 22. Under equilibrium conditions, hydrogen exists at room temperature as 75 per cent ortho and 25 per cent para. However, under equilibrium conditions at the boiling point of liquid hydrogen (20.3 K), the hydrogen is 99.7 per cent para and only 0.3 per cent ortho. Therefore, when hydrogen gas at room temperature is converted to liquid at 20.3 K, there is a tendency for 75 per cent of its molecules to undergo conversion to the para form. This conversion reaction is exothermic to the extent of 337.2 calories per mole converted, which may be compared with 215.9 calories per mole for the molal heat of vaporization. The ratio of these two values is 1.562; hence, 1.562 lb of hydrogen will be evaporated for each pound of ortho hydrogen that converts to the para form in the liquid state. This process accounts for the evaporation loss that occurs in storage of liquid hydrogen under atmospheric pressure, despite ideal conditions with respect to heat-leakage.

(b) Analysis of the hydrogen losses in a 100-lb capacity storage vessel similar to the design shown by Fig. 20, has been made by Dr. H. L. Johnston (Reference 7). The results of the analysis are plotted as Fig. 23. Based on the assumptions made, it would theoretically be possible to fill a 750-liter storage container in about 38 hours of plant operation at 25 liters per hr, and 4900 liters would be accumulated in 400 hours of operation. These figures do not include pre-cooling time requirements. If the ortho-para equilibrium were changed from the normal 75 per cent ortho to

50 per cent ortho by conversion in the liquefier prior to storage, the 750 liters could be accumulated in 34 hours and 6600 liters would accumulate in 400 hours of operation. Consequently, it is seen that the ortho-para conversion at the liquefier assumed above improves the storage characteristics of hydrogen by a considerable amount. An equilibrium of 50 per cent ortho could be attained by catalysis at the liquid nitrogen temperature.

(c) An analysis of the storage losses with surface catalysis assumed has also been made by Dr. H. L. Johnston (Reference 7). Rate units for the catalytic effect of various construction materials are reproduced from test data obtained at Ohio State University. The results of tests on cumulative evaporation loss resulting from surface catalysts are plotted as Fig. 24. For comparison, the accumulative volume at 400 hour (75 per cent ortho composition) for a stainless tank would be 4840 liters as compared to 4874 liters for noncatalyzed storage; the difference is small; therefore, the use of stainless steel or Monel in tank construction would appear to be optional. The most catalytic material listed is iron oxide which is roughly 4 times as active as stainless steel.

### 5 Design Criteria Summary

(a) The major factors to be considered in the storage of liquid hydrogen may be summarized as follows:

(1) Preferred materials for liquefied-gas storage vessels are the austenitic 18-8 stainless steels, Monel, Inconel, nickel, and the low-carbon, high-nickel steels.

(2) The use of powdery insulators such as Santocel, cork, and charcoal in an evacuated space is better than vacuum alone. The vacuum jacket pressure should be kept below about 100 microns for most efficient operation.

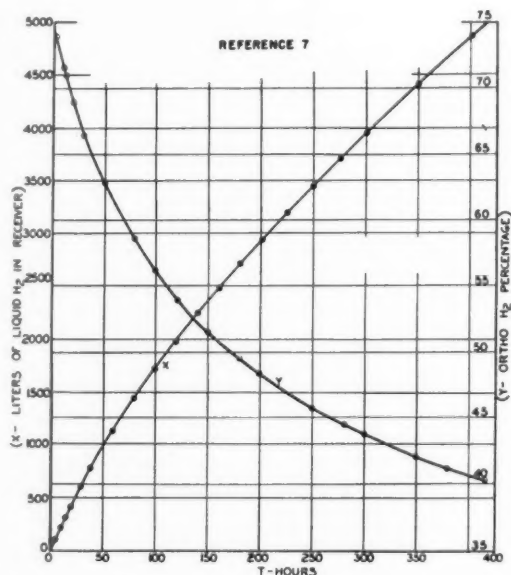


FIG. 23 CALCULATED LIQUID HYDROGEN ACCUMULATION AND ORTHO-PARA COMPOSITION VS. TIME FOR INPUT RATE OF 25 LITERS PER HR

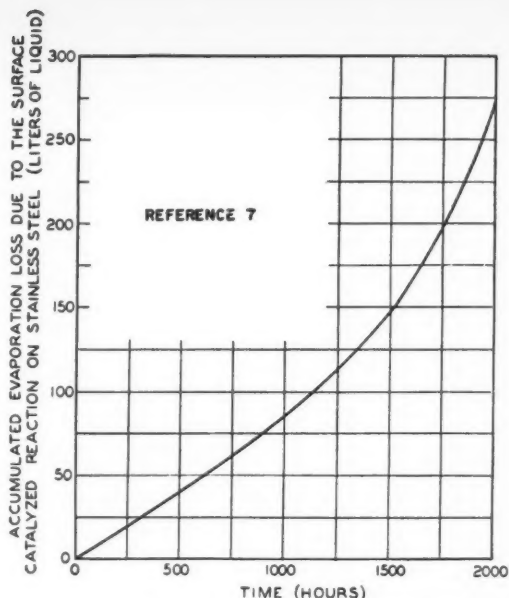


FIG. 24 CATALYZED EVAPORATION OF LIQUID HYDROGEN DUE TO SURFACE REACTION IN A STAINLESS STEEL STORAGE VESSEL

(3) Introduction of a radiation shield in the form of a liquid-nitrogen jacket or a liquid-nitrogen-cooled shield will also reduce the heat flow through the jacket considerably.

(4) An additional means of storing liquefied gases for long periods is to provide refrigeration equal to the heat influx (and any internal heat generation).

### B HANDLING AND SAFETY PRECAUTIONS

#### 1 Liquid Hydrogen

(a) On the whole, liquid hydrogen is less hazardous than high-pressure gaseous hydrogen, and it may, in fact, be regarded as a highly volatile gasoline. The primary precaution to be taken is prevention of air leakage into containers filled with liquid hydrogen. The air will freeze and either coat on the cold wall above the liquid hydrogen or sink to the bottom of the container. The solid air and liquid-hydrogen combination will not explode spontaneously; it requires a flame or spark. It should be mentioned however, that a possible source of such a spark is the breaking of a crystal of solid air or oxygen.

(b) Under normal circumstances, there is not much danger of an explosion (in the gaseous phase) because of the evaporation of liquid hydrogen in well-insulated containers. The hydrogen can evaporate only as fast as it receives heat from the outside or from its own ortho-para conversion. The vaporization rate will usually be rather low, and no hazard is normally present if good ventilation is provided. As a precaution against air entry, the vapors should be passed through a relief valve or mercury trap or provision should be made for the effluent vapors to re-enter the low pressure side of the liquefaction cycle.

(c) In storing liquid hydrogen, proper precautions

should be taken to see that the hydrogen vapor is vented off to prevent a tank pressure build-up in a safe manner so as not to create an atmospheric explosion hazard or allowing air entry into the vessel. In the event a storage vessel is damaged in use so that its contents are spilled, there is not apt to be an explosion unless a flame or spark is present. The hydrogen should be permitted to evaporate undisturbed, since it will evaporate very rapidly and will rise quickly. If it ignites, it will burn itself out and can be partially controlled with a carbon-dioxide fire extinguisher. However, the  $\text{CO}_2$  may be converted to  $\text{CO}$  by reaction with the heated hydrogen. Therefore, consideration should be given to the possible toxic atmosphere which could result.

(d) In a high-pressure system, there is some danger of the Dewar bursting if the Joule-Thomson valve is accidentally opened too quickly. This possibility may be eliminated by providing safety valves or bursting disks, which are set well below the burst pressure of the Dewar.

(e) In rocket piping systems, the high velocities used result in a high tolerance to heat influx because of the large mass-flow rates. The rate of bulk temperature rise in the fluid for various rates of flow has been computed and is shown in Fig. 25. It is assumed that the surface of this insulation has been sealed air-tight to prevent condensation of air in the jacket.

## 2 Gaseous Hydrogen

(a) The hazards in handling high-pressure hydrogen gas should not be lightly regarded. The explosive limits of hydrogen-air-mixtures are from about 5 per cent to 90 per cent hydrogen in air.

(b) One of the best precautions against explosions in an enclosed plant area is to provide a ventilating system which can handle the full flow of hydrogen in the system without permitting the lower explosive limit to be exceeded. However, even with this system, careful attention should be given to the elimination of any opportunity for the existence of local concentrations in excess of the lower explosive limit. One point in favor of hydrogen, from a hazard standpoint, is the fact that the gas rises very quickly and escapes from the area in which it is released. Ventilating hoods or vents and exhaust fans should be provided, however, at all places where possible failures of high pressure lines or unit containing gas could occur. Pressure tests should be very carefully made on all equipment before hydrogen is admitted into the system.

(c) In general, there are three definite steps which should be taken to provide for adequate protection against explosion: (1) Careful pressure and leak testing of all lines and equipment. (2) Provision of adequate ventilation. (3) Elimination of the likelihood of a flame or spark occurring in the hydrogen area. An accident can occur only if (1), (2), and (3) are neglected simultaneously.

(d) In regard to step (1) great care should be exercised in selecting piping, tubing, cryostat materials, etc., that will carry high-pressure hydrogen. A low-

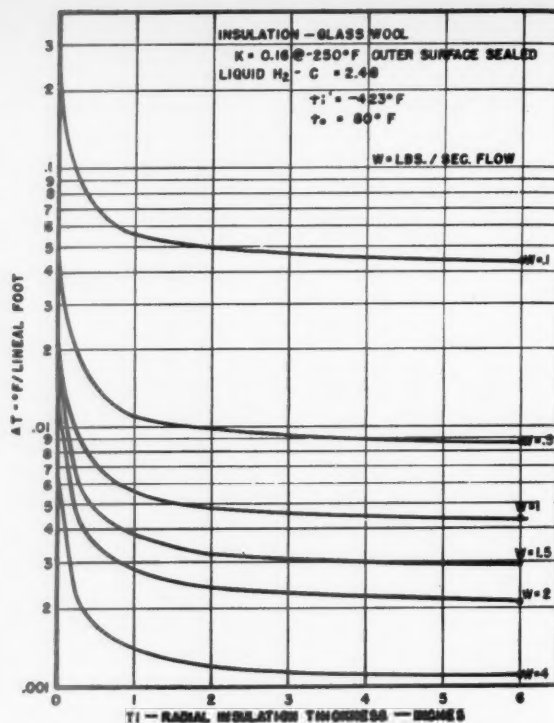


FIG. 25 TEMPERATURE RISE VS. INSULATION THICKNESS FOR 1-IN. TUBING

melting silver solder is recommended for high-pressure use, but the excessive heating of the metal to be joined should be carefully avoided. Monel and copper both have good impact properties down to liquid-hydrogen temperatures, and stainless steel is also satisfactory but somewhat more irregular. Impact characteristics of various materials at low temperatures are shown in Figs. 15, 16, and 17.

(e) In addition to providing adequate ventilation and circulation, a combustible alarm system, such as that made by the Mines Safety Appliance Company, should be arranged to analyze gas samples from all critical locations in the plant area. An alarm would then be sounded when the concentration of hydrogen at any station exceeded some value well below the lower explosive limit. This unit is also valuable in locating any leaks which might occur.

(f) To reduce the likelihood of a spark or flame occurring, the following recommendations should be observed in addition to the obvious precautions of not permitting open flames, smoking, etc. in the area: (1) All electrical wiring and machinery within the plant should be of the Class I, Group B type of construction in accordance with the Explosive Atmosphere Classification of the National Electrical Code. (2) All tools used within the immediate area while the plant is in operation should be of the spark-resistant, beryllium-copper type. (3) All belt-driven machinery should be equipped with spark-resistant belts. (4) All apparatus that carries

(Continued on page 338)



# Experimental Performance of WFNA-JP-3 Rocket Motors at Different Combustion Pressures<sup>1</sup>

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The performance and heat transfer rates for 500-lb thrust,  $L^* = 100$  in., rocket motors were determined experimentally as a function of mixture ratio at 300, 500, and 700-psia combustion pressures. In all cases the propellants used were white fuming nitric acid (WFNA) and jet engine fuel (JP-3). The maximum values of specific impulse were obtained at a mixture ratio of 4.5 and the values were 222, 235, and 246 for 300, 500, and 700-psia combustion pressure, respectively. Raising the combustion pressure from 300 psia to 700 psia increased the over-all heat transfer for the thrust cylinder from 1.3 to 2.3 Btu per sq in. per sec and that for the nozzle from 2.8 to 6.0 Btu per sq in. per sec.

## 1 Introduction

IN RECENT years considerable effort has been expended upon the development of rocket engines using white fuming nitric acid (WFNA) and jet engine fuel (JP-3) as the propellants. These propellants occupy an important place in rocket engine development because they can be made available in large quantities in the event of an emergency, and give adequate performance values for several types of application (1).<sup>4</sup>

Thermochemical calculations indicated that the specific impulse values for WFNA-JP-3 should not differ significantly from those for WFNA-aniline or WFNA-furfuryl alcohol mixed with aniline when these propellants are reacted at the same combustion pressure (2). Most of the experimental data for the WFNA-JP-3 propellants, reported in the unclassified literature, were obtained at combustion pressures close to 300 psia and confirm the conclusions based on thermochemical calculations.<sup>5</sup>

Currently the majority of liquid bipropellant rocket engines are designed for operation at combustion pressures around 300 psia. It has been known for some

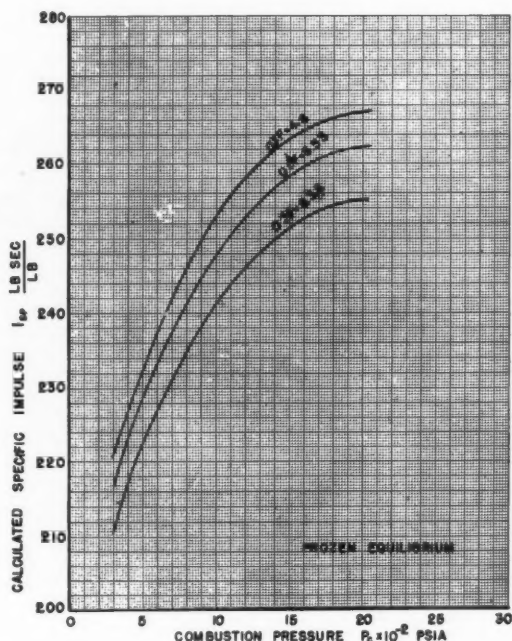


FIG. 1 AVERAGE THEORETICAL PERFORMANCE OF OCTANE OXIDIZED BY WFNA

time that increases in specific impulse should be obtainable by raising the combustion pressure. Fig. 1 presents calculated values of specific impulse as a function of combustion pressure for three different mixture ratios (oxidizer/fuel) for WFNA reacted with octane (4); the latter reaction is assumed to be representative of that for WFNA and JP-3. Fig. 2 presents calculated values of the ideal thrust coefficient  $C_F$  and the characteristic velocity  $C^*$  as functions of combustion pressure for the same propellants reacted at a mixture ratio (O/F) of 4.8. The results presented in Figs. 1 and 2 are based on frozen equilibrium. It is apparent from Figs. 1 and 2 that high combustion pressures offer the potentiality of achieving significant increases in performance from the WFNA-JP-3 propellants. There are, of course, numerous technical problems which must be solved before operation at the higher combustion pressures can be realized in a practical rocket engine (4), a major one being the adequate cooling of the thrust chamber and nozzle. To obtain an indication of the magnitude of the cooling problem to be expected at high combustion pressures, estimates of the heat transfer rates to the

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<sup>1</sup> This research program was conducted under the auspices of Project SQUID sponsored jointly by the Office of Naval Research and the Office of Air Research under Contract N6ori-105, and is presently to be continued by the sponsorship of the Office of Naval Research under Contract N7onr39418.

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<sup>3</sup> Research Assistant.

<sup>4</sup> Numbers in parentheses refer to the References on page 330.

<sup>5</sup> The authors regret that they cannot discuss the results other investigators obtained in the range 100 to 700 psia because those results are still classified for security reasons.

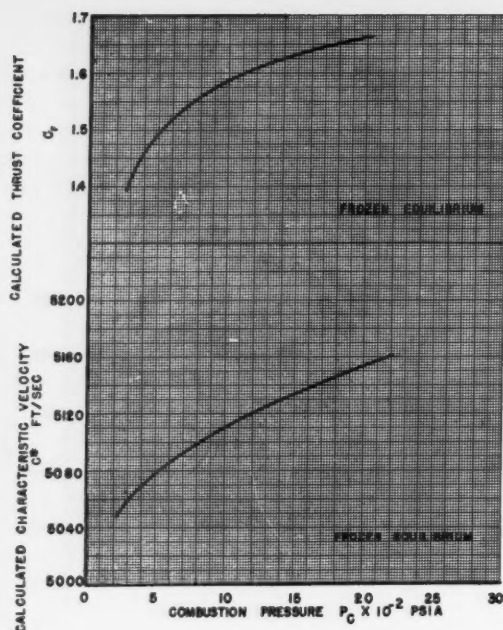


FIG. 2 VARIATION OF THEORETICAL  $C^*$  AND  $C_F$  WITH COMBUSTION PRESSURE FOR O/F EMPLOYING OCTANE OXIDIZED BY WFNA

thrust chamber and nozzle were made (3). The calculated results, which are approximations at best, are presented in Fig. 3. The set of curves labeled Humble, is based on the heat transfer coefficients presented in (5), and the curves, labeled McAdams, are based on (6). The estimates of heat transfer indicated that regenerative cooling of a stainless steel thrust chamber should be feasible for combustion pressures up to at least 1500 psia, and that regenerative cooling of a stainless steel nozzle would probably become marginal at combustion pressures of the order of 700 to 1000 psia.

Accordingly a research program, which is still in progress, was initiated at Purdue University to obtain experimental data concerning the obtainable specific impulses, the heat transfer rates, and general operating characteristics of geometrically similar rocket motors operating at high combustion pressures. The research program was originally sponsored by the Office of Naval Research and the Office of Air Research (Contract N6ori-104, Task Order 1, Project SQUID, Phase 7) and is now sponsored by the Office of Naval Research (Contract N7onr39418).

The present paper discusses the results for the experiments conducted at 300, 500, and 700 psia with three similar water-cooled rocket motors, each designed to produce 500 lb thrust, and each having an  $L^* = 100$  in. All of the experiments were conducted with WFNA and JP-3 as the propellants.

## 2 Description of the Experimental Rocket Motors

The general design features of all of the experimental rocket motors are illustrated by Fig. 4 which is a longitu-

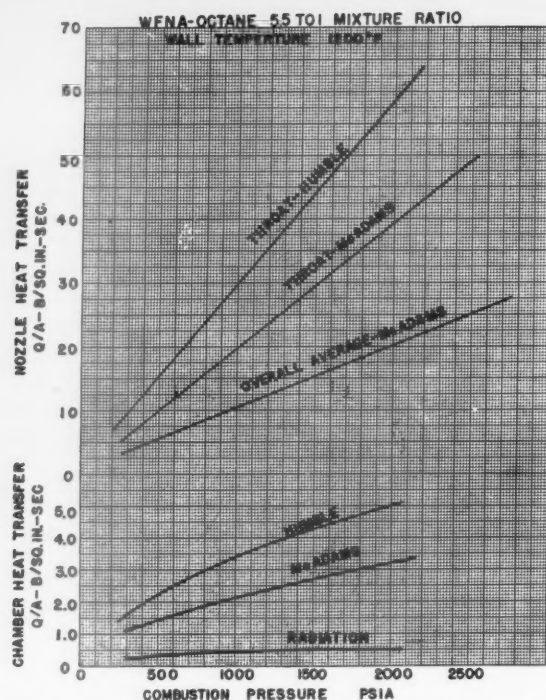


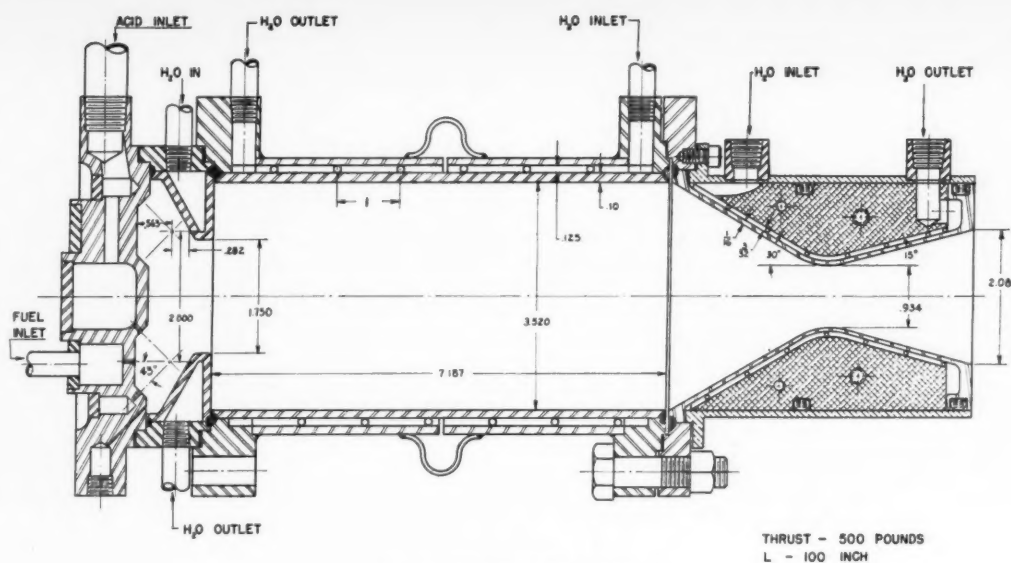
FIG. 3 CALCULATED HEAT TRANSFER RATES AS A FUNCTION OF CHAMBER PRESSURE FOR A 500-LB THRUST MOTOR

dinal cross section through the rocket motor employed in the experiments at 500 psia. In all cases the combustion chamber (thrust cylinder) was constructed from AISI Type 347 stainless steel. For the rocket motors employed for the experiments at 300-psia and 500-psia combustion pressures, the exhaust nozzle material was AISI Type 347 stainless steel. The rocket motor nozzle employed for the experiments at 700-psia combustion pressure was made from electrolytic copper; stainless steel nozzles were tested, but in all cases the throats became eroded when they were operated with mixture ratio close to the stoichiometric values.

In all cases the thrust chamber and the exhaust nozzle were water-cooled, each having its own cooling water circuit so that the over-all heat transfer rates for the thrust chamber and nozzle could be measured conveniently.

The coolant passages around the thrust chamber and nozzle were formed, by silver-soldering a stainless steel wire, wound helical fashion, around the outer surfaces of those components. The average velocity of the coolant (water) through the thrust chamber cooling passages was approximately 30 fps. The helical cooling passage around the nozzle had a variable pitch which increased the coolant velocity from 30 fps at the thrust cylinder end to approximately 80 fps in the vicinity of the nozzle throat and then reduced it again to 30 fps at the nozzle exit section. In the case of the rocket motor for the 700-psia combustion pressure experiments, a copper nozzle with the same type of cooling system was employed (8).

The rocket motors employed for the 500 and 700-psia experiments were equipped with 12-point triplet in-



jectors, and a water-cooled turbulence ring made from AISI Type 347 stainless steel was inserted between the injector face and the thrust cylinder. The same type of assembly was employed with the rocket motor employed for the 300-psia combustion pressure experiments, but in that case a 6-point triplet injector was utilized. The relative locations of the impingement points of the propellant streams from the upstream face of the turbulence ring are indicated in Fig. 4.

### 3 Preliminary Experiments to Establish Injection System

Since the complete research program was planned to conduct the experiments at 300, 500, 700, 1000, and 1500-psia combustion pressures, with the thrust and  $L^*$  held constant, it was important to employ an injection system which would exert a minimum influence upon the results, and which, within practical limits, would be geometrically similar for all of the experimental rocket motors. Moreover, it was deemed essential that the minimum performance obtained at each combustion pressure should not be less than 95 per cent of the theoretical performance, based on frozen equilibrium (see Figs. 1 and 2).

Because of the afore-mentioned requirements it was necessary to undertake some preliminary experiments, described later, for establishing a satisfactory injector configuration. The preliminary experiments conducted at 500-psia combustion pressure will be discussed.

A 6-point and a 12-point triplet injector were investigated with and without a turbulence ring.<sup>6</sup> In all cases the incorporation of a turbulence ring gave performance values in the range specified above. The authors are aware, however, that comparable performances are possible with other types of injection systems.

The objectives of the first or preliminary firing tests conducted at 500-psia combustion pressure were to establish the injection configuration to be employed for the main performance studies at that combustion pressure. In all of the experiments discussed in this paper, the ignition of the WFNA-JP-3 propellants was achieved by leading the flow of JP-3 with a small supply of either furfuryl alcohol or a furfuryl-aniline mixture. Table 1 presents some of the results obtained from the

<sup>6</sup> Acknowledgment is here given to R. B. Canright, Jet Propulsion Laboratory, California Institute of Technology, for furnishing information on the design of a turbulence ring investigated at IPL.

TABLE 1 PERFORMANCE DATA AT 500-PSIA COMBUSTION PRESSURE  
(Four methods of injection) (Propellants JP-3, WFNA)

Run no.	95	98 <sup>a</sup>	99	100 <sup>a</sup>
Injector no.	500-2:1-6	500-2:1-6TC	500-2:1-12	500-2:1-12TC
Combustion pressure, psia	495.7	532.9	536.7	525.5
Thrust, lb	493.4	524	510.2	513.5
Mixture ratio, O/F	5.04	4.05	4.88	5.10
Specific impulse, sec	208.9	227.0	214.8	228.4
Characteristic velocity, ft/sec	4607	4856	4755	4760
Average heat transfer rates:				
Nozzle, B/sec-in. <sup>2</sup>	2.46	3.60	2.78	4.03
Thrust cylinder, B/sec-in. <sup>2</sup>	1.23	2.19	1.71	2.18

<sup>a</sup> TC denotes carbon turbulence ring was used.



preliminary tests made at 500-psia combustion pressure.

It is apparent from Table 1 that both the 6-point and 12-point injectors gave larger values of specific impulse when employed in conjunction with a turbulence ring. The specific impulse values obtained with the turbulence ring did not differ materially from the calculated values for WFNA-JP-3 based on frozen equilibrium.

Examination of the interior surfaces of the rocket motor after a run, employing a turbulence ring, showed that the walls of the thrust chamber and nozzle were covered with a thin, uniform layer of porous carbon. Measurements indicated that the thickness of the carbon deposit varied from 0.020 in. to 0.027 in. At all combustion pressures it was found that if the turbulence ring was not employed, the carbon deposits were non-uniform and heavier, and formed in a definite pattern corresponding to the locations of the injector impingement points. Visual examinations of the turbulence ring after a large number of firing runs of 40 to 90 sec duration showed in all cases that its upstream surfaces remained clean, shiny, and free of carbon, while the

downstream surfaces were usually covered with a thin, smooth deposit of carbon. It is inferred that a substantial portion of the propellant mixture is in the liquid phase upstream to the turbulence ring.

#### 4 Performance Obtained at Different Combustion Pressures (7, 8)

Fig. 5 presents the experimental values of the specific impulse  $I_{sp}$ , the thrust coefficient  $C_F$ , and the characteristic velocity  $C^*$ , as a function of the mixture ratio (oxidizer/fuel by weight) for the WFNA-JP-3 propellants at 300-psia combustion pressure. The data were obtained with a 500-lb thrust, water-cooled rocket motor, having an  $L^* = 100$  in. The injection system comprised a 6-point triplet injector coating with a water-cooled turbulence ring.

Fig. 6 presents the values of specific impulse presented in Fig. 5 corrected for the heat transferred to the coolant. It is seen that the corrected values of  $I_{sp}$  are not significantly different from those computed upon the basis of frozen equilibrium.

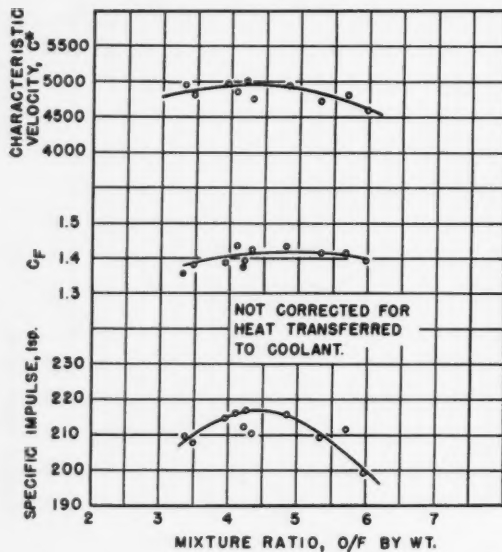


FIG. 5 PERFORMANCE VS. MIXTURE RATIO FOR WFNA-JP-3 AT 300-PSIA CHAMBER PRESSURE

(Thrust, 500 lb;  $L^*$ , 100 in.; injector, triplet type with 6 impingement points used in conjunction with a turbulence ring.)

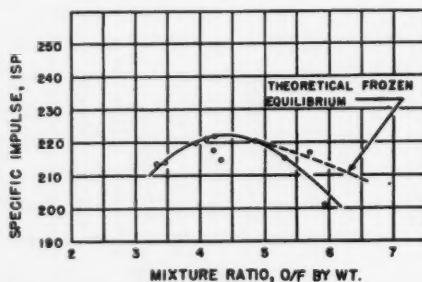


FIG. 6 SPECIFIC IMPULSE CORRECTED FOR HEAT TRANSFERRED TO COOLANT VS. MIXTURE RATIO FOR WFNA-JP-3 AT 300-PSIA CHAMBER PRESSURE

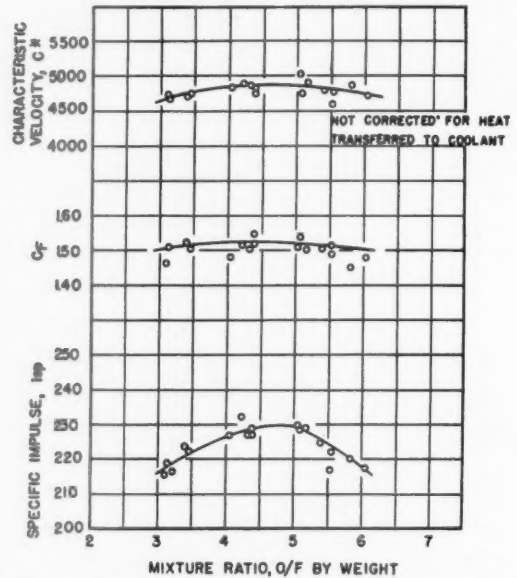


FIG. 7 PERFORMANCE VS. MIXTURE RATIO FOR WFNA-JP-3 AT 500-PSIA CHAMBER PRESSURE

(Thrust, 500 lb;  $L^*$ , 100 in.; injector, triplet type with 12 impingement points used in conjunction with a turbulence ring.)

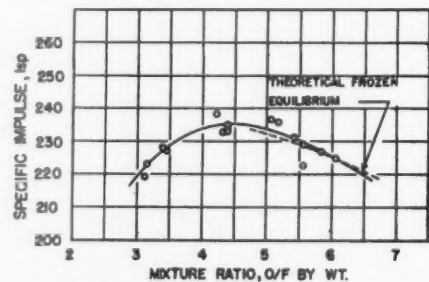


FIG. 8 SPECIFIC IMPULSE CORRECTED FOR HEAT TRANSFERRED TO COOLANT VS. MIXTURE RATIO FOR WFNA-JP-3 AT 500-PSIA CHAMBER PRESSURE



Figs. 7 and 9 present the experimental values of  $I_{sp}$ ,  $C_F$ , and  $C^*$  for the 500 and 700-psia combustion pressures, respectively, plotted as a function of the mixture ratio. Figs. 8 and 10 present the corresponding values of  $I_{sp}$  corrected for heat transfer; for comparison, the curve representing the theoretical values of  $I_{sp}$  based on frozen equilibrium are indicated in Figs. 6, 8, and 10.

Table 2 compares the maximum values of  $I_{sp}$  obtained for the three different combustion pressures. It is seen that in all cases the values of  $I_{sp}$  corrected for heat transfer are not significantly different from the calculated values based on frozen equilibrium. The last column in Table 2 presents the ratio of the maximum values of  $I_{sp}$  obtained at 500 and 700-psia combustion pressures to that obtained at 300-psia combustion pressure.

## 5 Heat Transfer to Thrust Cylinder and Nozzle

The accurate measurement of the heat transfer from combustion gases to the metal walls of a rocket motor is difficult to accomplish with precision. This is partic-

TABLE 2 COMPARISON OF THE MAXIMUM VALUES OF SPECIFIC IMPULSE FOR WFNA AND JP-3 AT DIFFERENT COMBUSTION PRESSURES

(Obtained from the three different rocket motors each designed for 500-lb thrust and  $L^* = 100$  in.)

Combustion pressure (psia)	Calculated values <sup>c</sup>		Experimental values		Experimental values		$(I_{sp})_{500} / (I_{sp})_{300}^d$
	$I_{sp}$ (sec)	O/F ratio	$I_{sp}$ (sec)	O/F ratio	$I_{sp}^d$ (sec)	O/F ratio	
300 <sup>a</sup>	222	4.5	217	4.5	222	4.5	1.0
500 <sup>b</sup>	234	4.5	230	4.5	235	4.5	1.058
700 <sup>b</sup>	245	4.5	240	4.5	246	4.5	1.108

<sup>a</sup> 6-point triplet injector and turbulence ring.

<sup>b</sup> 12-point triplet injector and turbulence ring.

<sup>c</sup> Based on frozen equilibrium according to Ref. 2.

<sup>d</sup> Corrected for heat transferred to cooling water.

ularly the case where the propellant combination tends to deposit carbon on the inner walls of the thrust cylinder and nozzle during a firing test. As is to be expected, the carbon deposits influence the heat transfer to the cooling water. The effect of "carboning-up" is illustrated in Fig. 11, which is a typical record of the temperature of the cooling water leaving the nozzle cooling jacket as a function of time; the particular curve applies to one of the experiments at 500 psia.

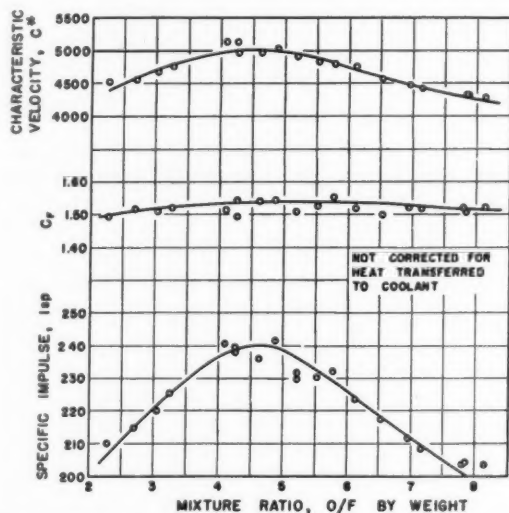


FIG. 9 PERFORMANCE VS. MIXTURE RATIO FOR WFNA-JP-3 AT 700-PSIA CHAMBER PRESSURE

(Thrust, 500 lb;  $L^*$ , 100 in.; injector, triplet type—two acid streams impinging on one fuel stream—with 12 impingement points used in conjunction with a turbulence ring.)

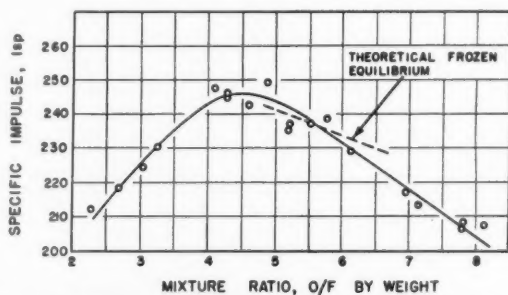


FIG. 10 SPECIFIC IMPULSE CORRECTED FOR HEAT TRANSFERRED TO COOLANT VS. MIXTURE RATIO FOR WFNA-JP-3 AT 700-PSIA CHAMBER PRESSURE

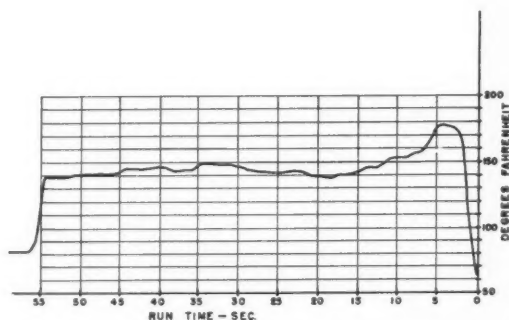


FIG. 11 NOZZLE COOLANT OUTLET TEMPERATURE. RUN NO. 99

The experiments indicate that it takes several seconds of running time for the heat transfer to reach a quasi-steady-state condition. They also indicate that with a clean, uncarboned motor, the heat transfer rates may reach peak values that are approximately 1.3 to 1.5 times as large as the reported quasi-steady-state values.

The heat transfer values reported herein are based on temperature readings taken over a period of at least 20 sec following the time when the cooling water temperatures reached substantially steady-state values.

Figs. 12, 13, and 14 present the experimental values of over-all heat transfer, as a function of O/F ratio, for the thrust chamber and nozzle, for the 300-psia, 500-psia, and 700-psia experiments, respectively. The figures also present the over-all values of heat transfer for the complete rocket motor.

The values of heat transfer computed by Ref. 3, summarized in Fig. 3, were for a 500-lb thrust rocket motor having a characteristic length  $L^* = 100$  in., burning WFNA-octane propellants. The aspect ratio for that rocket motor (length/diam) was two. The afore-

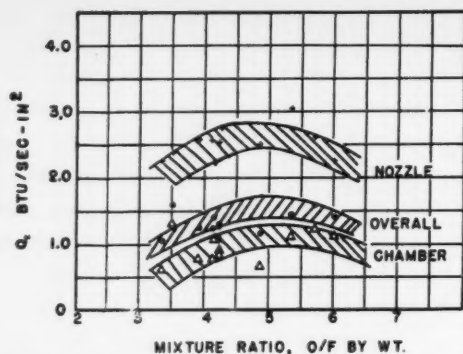


FIG. 12 HEAT TRANSFER VS. MIXTURE RATIO FOR WFNA-JP-3 AT 300-PSIA CHAMBER PRESSURE

(Thrust, 500 lb;  $L^*$ , 100 in.; nozzle heat transfer area, 38.1 in.<sup>2</sup>; chamber heat transfer area, 113.5 in.<sup>2</sup>; injector, triplet type with 6 impingement points used in conjunction with a turbulence ring.)

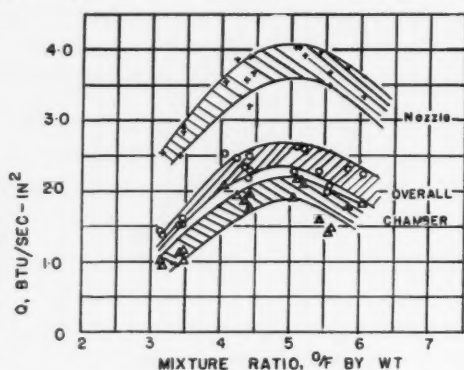


FIG. 13 HEAT TRANSFER VS. MIXTURE RATIO FOR WFNA-JP-3 AT 500-PSIA CHAMBER PRESSURE

(Thrust, 500 lb;  $L^*$ , 100 in.; nozzle heat transfer area, 29.75 in.<sup>2</sup>; chamber heat transfer area, 79.48 in.<sup>2</sup>; injector, triplet type with 12 impingement points used in conjunction with a turbulence ring.)

mentioned specifications are identical to those for the experimental rocket motors employed in these experiments. Consequently, a direct comparison of the calculated and experimentally determined heat transfer rates was possible. That comparison is presented in Table 3.

Referring first to the measured values of heat transfer, it is seen that for the thrust cylinder the ratio of the heat transfer at 500 psia and 700 psia to that at 300

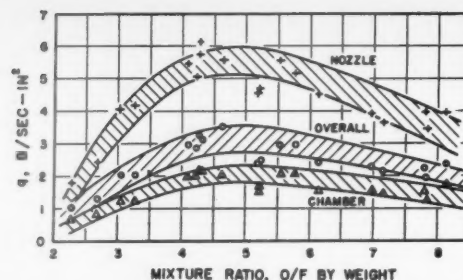


FIG. 14 HEAT TRANSFER VS. MIXTURE RATIO FOR WFNA-JP 3 AT 700-PSIA CHAMBER PRESSURE

(Thrust, 500 lb;  $L^*$ , 100 in.; nozzle heat transfer area, 25.22 in.<sup>2</sup>; chamber heat transfer area, 63.73 in.<sup>2</sup>; injector, triplet type with 12 impingement points used in conjunction with a turbulence ring.)

psia is 1.69 and 1.77, respectively. The corresponding ratios of the combustion pressures are 1.67 and 2.33, respectively. For the nozzle, the corresponding heat transfer ratios are 1.43 and 2.14. The calculated values for the same heat transfer ratio are for the thrust cylinder 1.3 and 1.6 by the Humble equation, and 1.25 and 1.5 by the McAdams equation. For the nozzle, the heat transfer ratios are 1.5 and 2.0 by the McAdams equation.

As pointed out earlier in this section, the quasi-steady-state measured values are less than the values obtained before any carbon is deposited on the metal walls of the thrust cylinder and nozzle. It is to be expected, therefore, that the calculated values of heat transfer, which are based on clean metal surfaces, should be larger than the measured values. Also, in view of the assumptions which were required to make the theoretical calculations, one cannot expect more than approximate agreement between the calculated and measured values based on clean metal walls. Both the calculated and measured values indicate that the heat transfer increases at a somewhat slower rate than the combustion pressure. The comparison indicates that the calculated values of heat transfer will be of the right order of magnitude for clean metal walls if the results obtained by Ref. 3 with the Humble equation are used for predicting the heat transfer for the thrust cylinder, and their results using the McAdams equation are employed for the over-all heat transfer in the nozzle.

TABLE 3 COMPARISON OF MEASURED AND CALCULATED VALUES OF THE MAXIMUM HEAT TRANSFER RATES FOR WFNA AND JP-3

(Obtained from three different rocket motors each designed for 500-lb thrust and  $L^* = 100$  in.)

Heat transfer rates in BTU per sq. in. per sec														Combustion pressure ratio
Combustion pressure (psia)	Measured values <sup>a</sup>						Calculated values <sup>b</sup>							
	Thrust cylinder		Nozzle		Rocket motor		Thrust cylinder		Nozzle		McAdams eq Ratio,	McAdams eq Ratio,	McAdams eq Ratio,	
	Ratio,		Ratio,		Ratio,		Ratio,		Ratio,					
	$q_c$	$\frac{q_c}{(q_c)_{300}}$	$q_n$	$\frac{q_n}{(q_n)_{300}}$	$q_0$	$\frac{q_0}{(q_0)_{300}}$	$q_c$	$\frac{q_c}{(q_c)_{300}}$	$q_c$	$\frac{q_c}{(q_c)_{300}}$				
300 <sup>c</sup>	1.3	1.00	2.8	1.00	1.7	1.00	1.6	1.00	1.1	1.00	4.0	1.00	1.00	
500 <sup>d</sup>	2.2	1.69	4.0	1.43	2.6	1.53	2.2	1.38	1.4	1.27	6.0	1.50	1.67	
700 <sup>d</sup>	2.3	1.77	6.0	2.14	3.5	2.06	2.7	1.69	1.7	1.55	8.0	2.00	2.33	

<sup>a</sup> Based on quasi-steady-state values after at least 20 sec of running.

<sup>b</sup> Taken from Ref. 3.

<sup>c</sup> 6-point triplet injector and turbulence ring.

<sup>d</sup> 12-point triplet injector and turbulence ring.

The above conclusions need additional experimental evidence before they can be regarded as being valid.

## 6 Description of Apparatus

The general arrangement of the test apparatus was quite conventional, so that it is unnecessary to present a detailed description of all of its components.

Fig. 15 is a schematic drawing illustrating the flow control and pressurizing system. Nitrogen from commercial bottles was used as the pressurizing medium. The nitrogen supply was divided into two separate pressurizing lines; one for the oxidizer (WFNA) flow system and the other for the fuel (JP-3) flow system. The pressurizing system for each propellant tank incorporated the same regulating devices: a dome pressure regulator followed by a solenoid valve. The pressures on the dome pressure regulators were adjusted from the control room by means of pressurizing-relief regulators.

Oxidizer and fuel were supplied to the rocket motor from two similar tanks, each constructed from AISI Type 347 stainless steel and hydrostatically tested to 6000 psi. Each tank has a capacity of 3.5 cu ft. All of the propellant lines are 1-in. diam AISI Type 347 stainless steel tubing, and all connections were made by means of flared tube fittings incorporating teflon gaskets. Filters were installed between the shut-off valves and the flowmeters. From the filters the propellants passed through the flowmeters and then to the rocket motor.

## 7 General Test Procedure

The diameter of the nozzle throat was measured at 0, 45, 90, and 135 deg before and after each run, and the

average of the measurements was employed in computing its cross-sectional area. The average difference in the throat area, determined in the afore-mentioned manner, before and after any run was 0.4 per cent approximately.

The thrust of the rocket motor was measured by two independent methods; electrically with a reluctance-type thrust ring, and mechanically with an air-loaded diaphragm-type device; the two instruments were connected in series through a lever arm arrangement. The electrical output from the thrust ring was recorded on an automatic balancing, recording potentiometer, and the pressure output from the air-loaded diaphragm-type unit was recorded on a strip chart pressure recorder. The electrical readings were used in the calculations (7).

The combustion pressure was measured with a reluctance-type pressure transducer and recorded simultaneously on a single-point automatic-balancing potentiometer, and an oscillograph; it was also indicated by a milliammeter on the control panel. The fuel and oxidizer injector pressures were measured, recorded, and indicated in a similar manner. Occasionally the propellant tank pressures were measured in a similar manner to determine the pressure drop in the propellant lines.

Four different methods were originally employed for measuring the fuel and oxidizer flow rates, the objective being to obtain information over a period of time for judging the accuracy and reliability of the different methods. Two of the methods gave readings of instantaneous flow rate, or its equivalent, and the other two furnished values of the average flow rate during a firing test. The instantaneous flow rates were measured (a) by means of sharp-edged orifices in conjunction

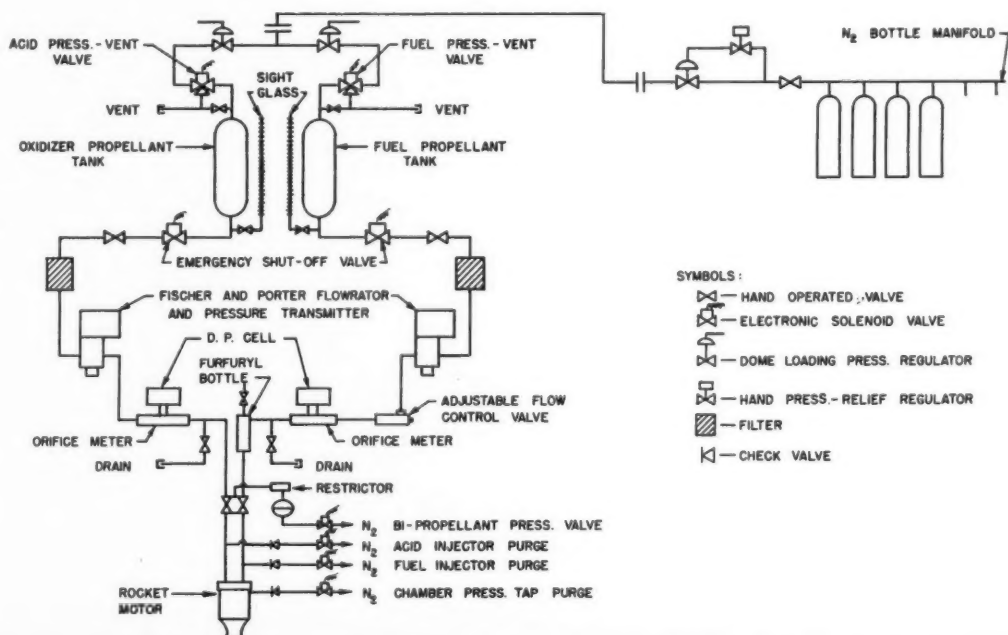


FIG. 15 SCHEMATIC DIAGRAM OF FLOW CONTROL AND PRESSURIZING SYSTEM

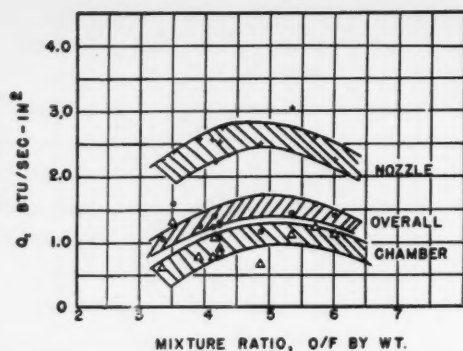


FIG. 12 HEAT TRANSFER VS. MIXTURE RATIO FOR WFNA-JP-3 AT 300-PSIA CHAMBER PRESSURE

(Thrust, 500 lb;  $L^*$ , 100 in.; nozzle heat transfer area, 38.1 in.<sup>2</sup>; chamber heat transfer area, 113.5 in.<sup>2</sup>; injector, triplet type with 6 impingement points used in conjunction with a turbulence ring.)

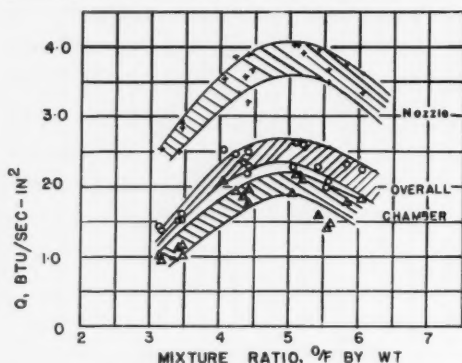


FIG. 13 HEAT TRANSFER VS. MIXTURE RATIO FOR WFNA-JP-3 AT 500-PSIA CHAMBER PRESSURE

(Thrust, 500 lb;  $L^*$ , 100 in.; nozzle heat transfer area, 29.75 in.<sup>2</sup>; chamber heat transfer area, 79.48 in.<sup>2</sup>; injector, triplet type with 12 impingement points used in conjunction with a turbulence ring.)

mentioned specifications are identical to those for the experimental rocket motors employed in these experiments. Consequently, a direct comparison of the calculated and experimentally determined heat transfer rates was possible. That comparison is presented in Table 3.

Referring first to the measured values of heat transfer, it is seen that for the thrust cylinder the ratio of the heat transfer at 500 psia and 700 psia to that at 300

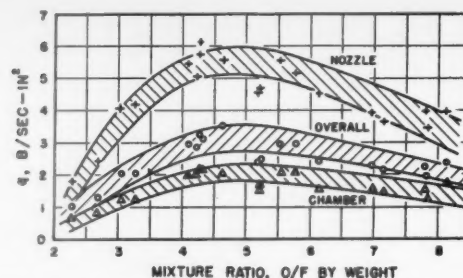


FIG. 14 HEAT TRANSFER VS. MIXTURE RATIO FOR WFNA-JP 3 AT 700-PSIA CHAMBER PRESSURE

(Thrust, 500 lb;  $L^*$ , 100 in.; nozzle heat transfer area, 25.22 in.<sup>2</sup>; chamber heat transfer area, 63.73 in.<sup>2</sup>; injector, triplet type with 12 impingement points used in conjunction with a turbulence ring.)

psia is 1.69 and 1.77, respectively. The corresponding ratios of the combustion pressures are 1.67 and 2.33, respectively. For the nozzle, the corresponding heat transfer ratios are 1.43 and 2.14. The calculated values for the same heat transfer ratio are for the thrust cylinder 1.3 and 1.6 by the Humble equation, and 1.25 and 1.5 by the McAdams equation. For the nozzle, the heat transfer ratios are 1.5 and 2.0 by the McAdams equation.

As pointed out earlier in this section, the quasi-steady-state measured values are less than the values obtained before any carbon is deposited on the metal walls of the thrust cylinder and nozzle. It is to be expected, therefore, that the calculated values of heat transfer, which are based on clean metal surfaces, should be larger than the measured values. Also, in view of the assumptions which were required to make the theoretical calculations, one cannot expect more than approximate agreement between the calculated and measured values based on clean metal walls. Both the calculated and measured values indicate that the heat transfer increases at a somewhat slower rate than the combustion pressure. The comparison indicates that the calculated values of heat transfer will be of the right order of magnitude for clean metal walls if the results obtained by Ref. 3 with the Humble equation are used for predicting the heat transfer for the thrust cylinder, and their results using the McAdams equation are employed for the over-all heat transfer in the nozzle.

TABLE 3 COMPARISON OF MEASURED AND CALCULATED VALUES OF THE MAXIMUM HEAT TRANSFER RATES FOR WFNA AND JP-3

(Obtained from three different rocket motors each designed for 500-lb thrust and  $L^* = 100$  in.)

Heat transfer rates in BTU per sq in. per sec													
Combustion pressure (psia)	Measured values <sup>a</sup>						Calculated values <sup>b</sup>						Combustion pressure ratio
	Thrust cylinder		Nozzle		Rocket motor		Thrust cylinder		Nozzle				
	Ratio, $\frac{q_c}{q_c}$	Ratio, $\frac{q_n}{q_n}$	Ratio, $\frac{q_o}{q_o}$	Humble eq Ratio, $\frac{q_c}{(q_c)_{300}}$	McAdams eq Ratio, $\frac{q_c}{(q_c)_{300}}$	McAdams eq Ratio, $\frac{q_n}{(q_n)_{300}}$							
$q_c$	$q_n$	$q_o$	$q_c$	$q_n$	$q_o$								
300 <sup>c</sup>	1.3	1.00	2.8	1.00	1.7	1.00	1.6	1.00	1.1	1.00	4.0	1.00	1.00
500 <sup>d</sup>	2.2	1.69	4.0	1.43	2.6	1.53	2.2	1.38	1.4	1.27	6.0	1.50	1.67
700 <sup>d</sup>	2.3	1.77	6.0	2.14	3.5	2.06	2.7	1.69	1.7	1.55	8.0	2.00	2.33

<sup>a</sup> Based on quasi-steady-state values after at least 20 sec of running.

<sup>b</sup> Taken from Ref. 3.

<sup>c</sup> 6-point triplet injector and turbulence ring.

<sup>d</sup> 12-point triplet injector and turbulence ring.



The above conclusions need additional experimental evidence before they can be regarded as being valid.

## 6 Description of Apparatus

The general arrangement of the test apparatus was quite conventional, so that it is unnecessary to present a detailed description of all of its components.

Fig. 15 is a schematic drawing illustrating the flow control and pressurizing system. Nitrogen from commercial bottles was used as the pressurizing medium. The nitrogen supply was divided into two separate pressurizing lines; one for the oxidizer (WFNA) flow system and the other for the fuel (JP-3) flow system. The pressurizing system for each propellant tank incorporated the same regulating devices: a dome pressure regulator followed by a solenoid valve. The pressures on the dome pressure regulators were adjusted from the control room by means of pressurizing-relief regulators.

Oxidizer and fuel were supplied to the rocket motor from two similar tanks, each constructed from AISI Type 347 stainless steel and hydrostatically tested to 6000 psi. Each tank has a capacity of 3.5 cu ft. All of the propellant lines are 1-in. diam AISI Type 347 stainless steel tubing, and all connections were made by means of flared tube fittings incorporating teflon gaskets. Filters were installed between the shut-off valves and the flowmeters. From the filters the propellants passed through the flowmeters and then to the rocket motor.

## 7 General Test Procedure

The diameter of the nozzle throat was measured at 0, 45, 90, and 135 deg before and after each run, and the

average of the measurements was employed in computing its cross-sectional area. The average difference in the throat area, determined in the afore-mentioned manner, before and after any run was 0.4 per cent approximately.

The thrust of the rocket motor was measured by two independent methods; electrically with a reluctance-type thrust ring, and mechanically with an air-loaded diaphragm-type device; the two instruments were connected in series through a lever arm arrangement. The electrical output from the thrust ring was recorded on an automatic balancing, recording potentiometer, and the pressure output from the air-loaded diaphragm-type unit was recorded on a strip chart pressure recorder. The electrical readings were used in the calculations (7).

The combustion pressure was measured with a reluctance-type pressure transducer and recorded simultaneously on a single-point automatic-balancing potentiometer, and an oscillograph; it was also indicated by a milliammeter on the control panel. The fuel and oxidizer injector pressures were measured, recorded, and indicated in a similar manner. Occasionally the propellant tank pressures were measured in a similar manner to determine the pressure drop in the propellant lines.

Four different methods were originally employed for measuring the fuel and oxidizer flow rates, the objective being to obtain information over a period of time for judging the accuracy and reliability of the different methods. Two of the methods gave readings of instantaneous flow rate, or its equivalent, and the other two furnished values of the average flow rate during a firing test. The instantaneous flow rates were measured (a) by means of sharp-edged orifices in conjunction

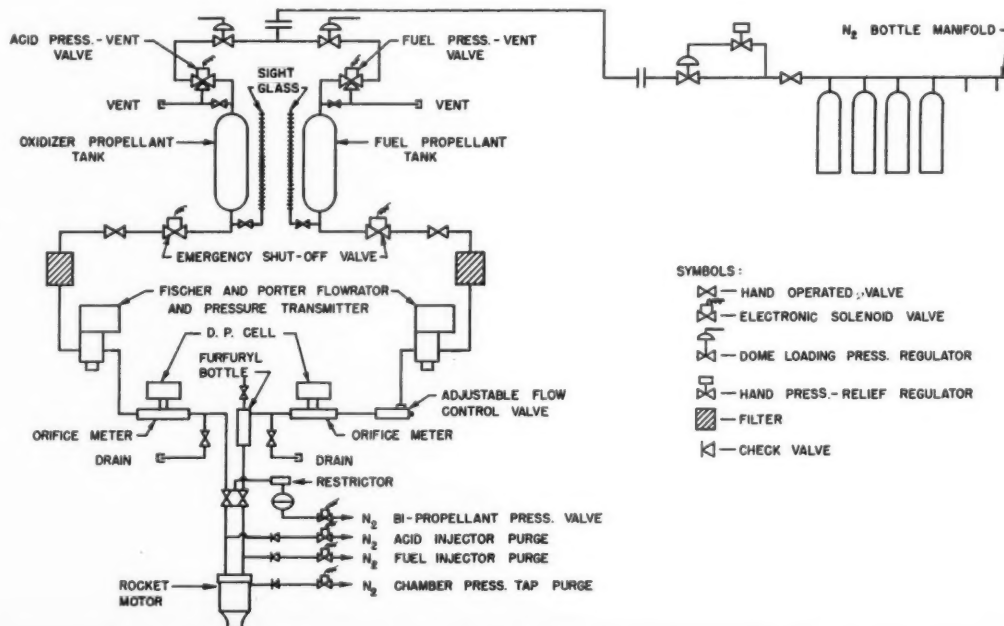


FIG. 15 SCHEMATIC DIAGRAM OF FLOW CONTROL AND PRESSURIZING SYSTEM

with a reluctance-type differential pressure transducer, and (b) by a rotometer equipped with a pressure transmitter. The electrical output from each transducer was recorded by an electronic automatic-balancing potentiometer and by an oscillograph. The output pressure from the rotometer was recorded on a pressure recorder and indicated on a mercury manometer. The most consistent readings were obtained from the sharp-edged orifices, and those readings were used in the calculations.

The total weight of each propellant consumed during a run was measured by a propellant-tank weighing system comprising a reluctance-type force ring transducer for measuring the tank weight. The electrical output of the transducer was indicated on an automatic-balancing potentiometer. In addition, the total weight of a propellant consumed during a run was calculated from the difference in liquid levels indicated by a sight glass attached to the propellant tank.

The specific gravities of the propellants were measured with a hydrometer and could be estimated to the fourth significant figure; the specific gravity measurements were taken before and after each run and the average of the two readings was employed in the calculations.

The flow rates of the cooling water to the thrust cylinder and to the nozzle cooling jackets were measured with sharp-edged orifice plates, the differential pressures being indicated on differential pressure gages. The orifice meters were calibrated at frequent intervals during the investigation, and the calibrations never differed by more than one per cent.

The temperatures of the cooling water entering and leaving the thrust chamber and nozzle cooling jackets were measured with iron-constantan thermocouples, and their readings recorded on an automatic-balancing, multipoint automatic potentiometer. Mixing chambers were installed at the entrance and exit to the cooling jackets and the thermocouples gave mixing-cup temperatures. The flow rates were adjusted to give temperature rises of the order of 100 deg F, and the temperature could be read to  $\pm 1$  deg F.

All of the individual measurements employed in calculating performance parameters were in error by less than 1 per cent. No analysis was made to determine the probable accuracy of the curves presented herein. It is deemed, however, that readings obtained from the curves should be in error by less than 2 per cent.

## 8 Conclusions

From the experiments reported herein, the following conclusions may be drawn:

(a) With a 12-point or a 6-point triplet injector and a turbulence ring in the experimental 500-lb rocket motors tested, the measured values of specific impulse  $I_{sp}$ , corrected for the heat transferred to the cooling water, were not materially different from the calculated values, based on frozen equilibrium.

(b) With the turbulence ring the heat fluxes for the thrust chamber and nozzle were increased to approximately the same values for both the 6- and 12-point injectors (see Table 1).

(c) Observation of the interior of the rocket motor after a run indicated that the main combustion occurs downstream of the turbulence ring, and that the actions of the latter were essentially those of a mixing device and possibly of a flameholder.

(d) The mixture ratio giving the values of the maximum specific impulse, corrected for heat transferred to the cooling water, was practically 4.5 to 1 at all combustion pressures.

(e) The maximum values of heat transfer at all combustion pressures, for both the thrust chamber and the nozzle, occurred at mixture ratios not significantly different from 5.0 to 1.

(f) The measured heat fluxes when burning a hydrocarbon fuel, such as JP-3, are influenced by the carbon being deposited on the walls of the rocket motor during the operation period—the heat transfer rate decreasing with the operating time. With a clean motor, heat transfer rates approximately  $1\frac{1}{2}$  times those corresponding to the reported steady state values may be expected.

## Acknowledgments

The authors hereby gratefully acknowledge the valuable assistance they received from their colleagues, D. E. Robison, Research Assistant, and R. E. Watters, Laboratory Assistant, Purdue University.

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# An Approximate Theory of Porous, Sweat, or Film Cooling With Reactive Fluids<sup>1</sup>

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Rannie's approximate theory of porous cooling for inert coolants is here extended to the case in which the coolant itself is reactive (for instance, the coolant is one of the propellants). The extension is possible with the help of drastic assumptions about the diffusions and the reactions of the gases. The theory can be applied to the case of film cooling, provided the liquid film is stable. A numerical application is developed.

## Nomenclature

$A$	= area of the cross section of the rocket motor
$c_F$	= friction coefficient of the boundary layer
$c_p$	= specific heat at constant pressure
$d$	= diameter of the rocket motor
$D$	= diffusion coefficient
$E$	= chemical energy of the fluid per unit weight
$h$	= enthalpy of the gases
$h_t$	= $h + E$ = total enthalpy
$H(\omega)$	= $E_c - E(\omega)$ = thermal effect of combustion of a mixture with oxidizer concentration $\omega$
$l$	= length of rocket motor
$m$	= rate of transfer of oxidizer
$Pr$	= $\mu c_p / \lambda$ = Prandtl number
$q$	= rate of heat transfer
$Q$	= $h - h_c + r_c$
$r$	= latent heat of vaporization of the coolant (zero when gaseous)
$Re$	= Reynolds number of the combustion gases in the rocket motor based on the diameter
$S$	= cylindrical cooled surface of the rocket motor
$Sc$	= $\mu / \rho D$ = Schmidt number
$T$	= temperature
$u$	= $x$ -component of velocity
$u_\tau$	= $(\tau_0 / \rho)^{1/2}$ = characteristic velocity of the boundary layer
$v$	= $y$ -component of velocity
$w$	= mass velocity
$W$	= total flow
$x$	= axial coordinate
$y$	= transversal coordinate
$y_\delta$	= constant related to the thickness of the laminar sublayer
$\delta$	= thickness of the laminar sublayer
$\eta$	= $\rho \omega / \rho_c \int^y (dy / \mu) =$ reduced ordinate
$\eta_\delta$	= reduced thickness of the laminar sublayer
$\lambda$	= thermal conductivity
$\mu$	= viscosity coefficient
$\rho$	= density
$\tau$	= shearing stress

$\Phi$  = auxiliary quantity defined by Equation [12]  
 $\omega$  = oxidizer specific concentration (weight of oxidizer per unit total weight)

## Subscripts

$c$  = coolant  
 $g$  = main gas stream  
 $0$  = boundary between gas and solid or liquid  
 $\delta$  = boundary between laminar and turbulent

## 1 Introduction

THE conventional regenerative cooling of rocket motors is subject to severe limitations in two cases: (a) When high energy propellants or high working pressures are used, since the process of cooling with a liquid stream the heated surface of a wall through its finite thickness has a limited effectiveness, and therefore there is a limit to the amount of heat that can be supplied to the wall without exceeding the prescribed safe temperature; (b) When the propellants due to their physical or chemical properties are unsuited for use in a regenerative cooling system.

On the other hand porous, sweat, and film cooling are free from limitation (a) because it is always possible with a sufficiently large supply of coolant to build a fluid film adequate to give the required protection to the solid wall; and they are free from limitation (b) because these types of cooling, taken together, are able to cover a wide range of physicochemical properties of the coolant. As a result of this better flexibility, porous, sweat, or film cooling might play an interesting role in the future development of rocket motors.

Rannie (1)<sup>3</sup> has developed a simplified theoretical treatment of the problem of porous cooling, with the purpose of correlating the mass flow of coolant through the porous wall and the temperature of the wall itself.<sup>4</sup> Rannie's theory is based on the following assumptions:

(a) The flow in the rocket can be divided into two regions: A central core, where the gases are those of combustion and are not affected by the cooling process, and a region adjacent to the wall where all the effects of the cooling process are confined, and which can be treated with the same simplifications as a conventional boundary layer.

(b) Starting in the axial direction from the section where the cooling begins, the central core and the bound-

<sup>3</sup> Numbers in parentheses refer to the References on page 338.

<sup>4</sup> Other theoretical treatments have been presented elsewhere (see e.g., Ref. 2) based on analogous assumptions. Rannie's theory seems, however, to be the most exhaustive.

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<sup>1</sup> For purposes of brevity, a distinction has been made in this paper between (a) porous cooling, i.e., cooling through a porous wall with a gas or a liquid vaporized before entering the wall, and (b) sweat cooling, where the coolant is liquid throughout the porous wall.

<sup>2</sup> Robert H. Goddard Professor of Jet Propulsion.



dary layer reach very soon a quasi-equilibrium condition where the axial derivatives of the different quantities involved can be practically neglected. The different quantities are therefore a function only of the distance from the wall within the boundary layer, and are constant in the central core.

(c) The relatively small velocity of the coolant coming from the wall does not affect appreciably the hydrodynamical condition of the boundary layer which, for sufficiently high Reynolds numbers, will be mainly turbulent except for a small laminar sublayer near the wall where the turbulent fluctuations are killed off by the presence of the solid boundary. Actually, there is no such sharp transition between the two conditions, and the discontinuity is only an approximation usually accepted when the Prandtl number is sufficiently close to unity.

(d) The physical properties of the coolant and of the combustion gases are the same and are independent of the temperature.

(e) The gas used as a coolant is inert with respect to the combustion gases.

Later Sloop has pointed out that Rannie's treatment can be applied to the sweat or film cooling problem,<sup>5</sup> where now the porous wall with the outcoming transversal flow of gaseous coolant is replaced by the liquid film boundary supplied continuously (either transversally or axially) with the outcoming transversal flow of vapor. All of the previously mentioned assumptions can be held provided:

(f) The liquid film is stable on the wall, so that its surface is smooth and gives to the gaseous laminar sublayer the same support as a sliding solid boundary.

In this case it is possible to use the results of Rannie's theory, slightly transformed, and to correlate theoretically the amount of coolant evaporated from the surface with the other data of the problem.

If condition (f) is not fulfilled, and the smoothness of the liquid film gives up under the unstabilizing action of the shearing stress between gas and liquid and of the gas turbulent fluctuations transmitted through the laminar sublayer, then two disturbing facts appear: First, the corrugations of the liquid film can have in turn a reaction on the laminar turbulent conditions in the boundary layer (in other words the film reacts in a different way than a solid boundary); and, second, the corrugations can grow so large in size that they are finally detached in drop form from the film and are carried away from it by the gaseous stream, without further contribution to the cooling mechanism (5). In this eventuality, since the porous cooling theory as applied to sweat or film cooling does not take into account this supplementary coolant consumption, the total coolant consumption is certainly larger (in some cases much larger) than the theoretical value. However, even in this case the theoretical prediction has an interesting meaning as a measure of the minimum possi-

ble value of the coolant consumption that could be obtained after stabilization of the film surface. Moreover, the difference between theoretical and actual coolant consumption or other related quantities, affords a measure of the effects of film instability.

The results of Rannie's computations check reasonably well with the experimental results of Duwez<sup>6</sup> (4). In the same way reasonable agreement can be expected in the film-cooling case, in the region where the film is stable. Despite the number of simplifying assumptions, the theory represents, therefore, a useful tool for predictions on these kinds of cooling when an inert coolant is used (assumption (e)).

Unfortunately, this is not the most interesting case for practical applications, since usually the coolant is represented by one of the propellants and is therefore capable of reacting chemically with the other propellant or with the products of combustion. The purpose of this paper is to extend Rannie's theory to this case, or more generally, to the case when the coolant can react with the combustion gases. The extension can only be done with the help of some additional simplifying assumptions. On the other hand, it will be shown that the quite restrictive assumption (d) can be dropped without introducing any essential complication in the treatment.

## 2 Supplementary Assumptions

As we have stated, we maintain all Rannie's assumptions except (d) and (e). About the effects of the reactions between coolant and combustion gases we make the following observations and assumptions. As the coolant gas is introduced or the coolant vapor is evaporated, it is diffused from the boundary toward the combustion gases. At the same time the combustion gases are diffused from the region external to the boundary layer toward the boundary, so that their relative concentration is different at different layers, varying from a minimum near the boundary to unity outside the boundary layer. This varying concentration has an important effect, since the combustion gases contain the molecular species which are able to react with the coolant. Suppose, for simplicity, that the coolant is a fuel, which therefore can react with the oxygen contained in the combustion gases in bound or free form. We make the following simplifying assumptions. These assumptions are quite crude, but drastic simplifications are needed if any answer must be obtained.

(g) The combustion gases are diffused as a whole. This is almost exact in the turbulent layer, since the diffusion in this case consists in macroscopic mixing; but it is certainly not exact in the laminar sublayer where the molecular diffusion rates are different for different molecular species. The consequence of this assumption is that when the coolant coincides with the fuel the mixture at every layer can be supposed to be derived from the mixing of certain amounts of fuel

<sup>5</sup> Computation on the same general line had already been developed by Shurman (Ref. 3).

<sup>6</sup> The same applies for the results of Ref. 2.



and oxidizer, and a single parameter, e.g., the oxidizer specific concentration, is sufficient to characterize the gross composition of the mixture.<sup>7</sup> The same property is assumed without further justification when the coolant is different from the fuel.

(h) The reaction times of the mixtures are short with respect to the other times involved, so that at every layer the mixture can be considered to reach immediately the final condition it would reach after a sufficiently long time. If the temperature, and, therefore, the oxidizer concentration are not too low, this final condition coincides with the equilibrium condition of the corresponding combustion gases. The consequence of this assumption is that the chemical composition of the mixture is known when the oxidizer specific concentration is known, and, therefore, the enthalpy of the mixture is a function of the oxidizer specific concentration and of the temperature. From the enthalpy  $h$  we can obtain the total enthalpy  $h_t$  if we add to  $h$  the potential energy  $E$  still contained in the combustion gases in chemical form. i.e., the heat that has not been released because of the incomplete oxidation. This chemical energy is a function only of the oxidizer specific concentration, and it is generally a maximum at the boundary and a minimum in the region external to the boundary layer.

(i) In the laminar sublayer the diffusion rate of the oxygen-carrying combustion gases is proportional to their total molecular concentration gradient, and the diffusion coefficient  $D$  is related to the viscosity coefficient and the density by the formula  $D = \mu/\rho \cdot Sc$ , where the Schmidt number  $Sc$  is assumed to be constant within the laminar sublayer. In fact, the diffusion rates follow more complicated laws, depending on the actual composition of the mixture, and only in the case in which all the components of the mixture have the same molecular weight is the aforesaid simple relation of proportionality exact. In order to obtain a further simplification, we assume that the molecular concentration is simply proportional to the oxidizer specific concentration. This proportionality would be exact only if the average molecular weight of the combustion products remained constant. Obviously the diffusion of the coolant vapor is reduced (assumption  $g$ ) to a complementary process, and does not need to be considered separately. About the assumption of constant  $Sc$ , it can be observed that actually this quantity is not far from unity, close to the Prandtl number, and its variations with temperature or composition are contained in small ranges.

(j) In the turbulent region the Reynolds analogy, which assumes that the same mechanism is responsible for the transfer of axial momentum and energy, can be extended also to the mass transport and, therefore, to the turbulent diffusion of chemical species.

<sup>7</sup> The sum of oxidizer specific concentration and of the fuel specific concentration is obviously one, and the latter could have been selected as parameter with equivalent results.

### 3 Relations in the Turbulent Region

Assumption (j) can be expressed more explicitly in the following way: When a fluid particle is displaced by the turbulent fluctuation from one layer to the other, it is displaced as a whole, carrying its axial momentum, its total energy content (more exactly its total enthalpy since the process takes place at constant pressure), and its chemical composition. There is, therefore, similarity between the fields of velocity, total enthalpy, and oxidizer concentration, and also between these fields and the rate at which momentum, energy, and concentration are transferred in the transversal direction. The similarity can be expressed by the following extended Reynolds formula

$$\frac{u_g - u_\delta}{\tau} = \frac{\omega_g - \omega_\delta}{m} = \frac{h_{tg} - h_{t\delta}}{q} \dots \dots \dots [1]$$

where  $\tau$  (turbulent stress),  $m$  and  $q$  represent, respectively, the rate at which momentum, concentration, and energy are transferred across a certain layer;  $u$ ,  $\omega$  and  $h_t$  are the axial velocity, the oxidizer specific concentration and the total enthalpy; and the subscripts  $g$ ,  $\delta$  indicate, respectively, the conditions of the main stream of gases outside the turbulent layer and the conditions at the idealized boundary between the turbulent layer and the laminar sublayer. Equation [1] can be applied to any layer in the turbulent region.

### 4 Relations in the Laminar Sublayer

The equations in the laminar region can be derived by a simple balance of mass, momentum, concentration, and energy in the  $y$ -direction, since by assumption (b) the quantities do not vary appreciably in the  $x$ -direction. Writing first the balance of mass, the mass flow per unit area and time at any distance  $y$  must be equal to the mass flow at  $y = 0$ , that is, at the solid or liquid surface where the gaseous boundary layer begins. Therefore, representing with subscript zero the quantities at  $y = 0$ , we have at any section

$$\rho v = \rho_0 v_0 \dots \dots \dots [2]$$

$\rho$  being the density and  $v$  the transversal velocity.

Between the two sections there is a variation of axial velocity and, therefore, of axial momentum, which can only be produced by the difference of viscous stresses on the two sections, since the other forces are assumed to be negligible. Therefore we obtain:

$$\tau - \tau_0 = \mu \frac{\partial u}{\partial y} - \left( \mu \frac{\partial u}{\partial y} \right)_0 = \rho v (u - u_0) \dots \dots \dots [3]$$

Considering now the balance of oxidizer, between the same two sections, we see that there is no exchange of oxidizer at  $y = 0$ , where there is only a flow of coolant, so that there must be balance at the other section between the mols of oxidizer carried by convection which by assumption is proportional to  $v\omega$  and the mols transferred in the opposite direction by diffusion, proportional to  $Dd\omega/dy$  with the same coefficient of proportionality. Remembering the introduction of the Schmidt number

(assumption *i*), we can write the balance of oxidizer under the form

$$m = \frac{\mu}{Sc} \frac{d\omega}{dy} = \rho v \omega \dots \dots \dots [4]$$

The coolant balance is automatically satisfied (assumption (*g*)).

Finally, let us consider the coolant at its supply temperature  $T_c$ , that is, for sweat or porous cooling, the temperature of the coolant before entering the porous wall and for film cooling, a temperature close to its boiling temperature under the prevailing pressure, the heat necessary to reach this temperature having been supplied in the first period of existence of the film after entering the combustion chamber. If we indicate with subscript *c* the quantities corresponding to this temperature, we see that the increase of total enthalpy per unit mass at ordinate *y* is given by  $h_t - h_{t_c} + r_c$ , where  $r_c$  is the latent heat of vaporization at temperature  $T_c$  ( $r_c = 0$  if the coolant is gaseous),  $h_{t_c}$  is the total enthalpy of the vapor at temperature  $T_c$ , and  $h_t$  is the total enthalpy of the mixture at ordinate *y*, which is, by assumption *h*, completely defined when *T* and  $\omega$  are known. On the other hand, the only supply of energy from outside is the heat transferred by conduction at ordinate *y*, that is  $\lambda dT/dy$  per unit area and time. In conclusion, introducing the Prandtl number  $Pr = \mu c_p / \lambda$ , we find that the balance of energy can be written under the form

$$q = \frac{\mu}{Pr} \frac{dh}{dy} = \rho v (h_t - h_{t_c} + r_c) = \rho v [h - h_c + r_c - H(\omega)] \dots \dots \dots [5]$$

where we have introduced the *thermal effect*

$$H(\omega) = E_c - E(\omega) = E(0) - E(\omega) \dots \dots \dots [6]$$

corresponding to the combustion of a mixture of fuel vapor and oxidizer with oxidizer concentration  $\omega$ . Obviously  $H(\omega)$  is an essentially positive quantity.

The integration of Equations [3], [4], and [5] with constant *Sc* and *Pr* is immediate. Introducing for brevity the reduced ordinate

$$\eta = \int_0^y \frac{\rho v}{\mu} dy = \rho v_0 \int_0^y \frac{dy}{\mu}$$

where Equation [2] has been used, we find that the integrals of Equations [3] and [4] are

$$u - u_0 = C_1(e^\eta - 1); \quad \omega = C_2 e^{Sc \cdot \eta}$$

The integration constants  $C_1, C_2$  can be expressed for instance through the values  $u_\delta$  and  $\omega_\delta$  at the limit  $y = \delta$  of the laminar sublayer. Writing

$$\eta_\delta = \rho v_0 \int_0^\delta \frac{dy}{\mu} = \frac{\rho v_0 \delta}{\mu_m} \dots \dots \dots [7]$$

$\mu_m$  representing an average value of  $\mu$  in the laminar sublayer, we have

$$u - u_0 = (u_\delta - u_0) \frac{e^\eta - 1}{e^{\eta_\delta} - 1}; \quad \omega = \omega_\delta e^{Sc(\eta - \eta_\delta)} \dots \dots [8]$$

At  $\eta = 0$  we see that the oxidizer concentration is  $\omega_0 = \omega_\delta \exp(-Sc \cdot \eta_\delta)$  and is, therefore, different from zero.

Coming now to the energy Equation [5], we see that  $\omega$  and, therefore,  $H(\omega)$  being known functions of  $\eta$ , the equation is a linear differential equation between  $h$  and the reduced ordinate  $\eta$ .

Introducing for brevity

$$Q = h - h_c + r_c \dots \dots \dots [9]$$

we see that the integral of Equation [5] is

$$Q = e^{Pr \cdot \eta} [Q_0 - Pr \int_0^\eta H(\omega(\eta_1)) e^{-Pr \cdot \eta_1} d\eta_1] \dots \dots [10]$$

where  $\eta_1$  is an integration variable,  $\omega(\eta_1)$  is given by the second Equation [8] with  $\eta = \eta_1$ , and  $Q_0$  is given by the value of [9] at  $y = 0$  and represents the total amount of heat transferred from the gaseous boundary layer to the porous wall or to the liquid film.

## 5 Turbulent-Laminar Junction and Solution of the Problem

Equation [3] applied at the external limit of the laminar sublayer gives

$$\tau_\delta = \left( \mu \frac{du}{dy} \right)_\delta + \rho v (u_\delta - u_0) = \rho v \left[ \left( \frac{du}{d\eta} \right)_\delta + u_\delta - u_0 \right] = \rho v \frac{u_\delta - u_0}{1 - e^{-\eta_\delta}}$$

the last step being obtained from the first Equation [8]. Moreover we have from Equations [4], [5]

$$m_\delta = \rho v \omega_\delta; \quad q_\delta = \rho v [Q_\delta - H(\omega_\delta)]$$

where  $Q_\delta$  is the value given by Equation [10] at  $\eta = \eta_\delta$ .

On the other hand in the present assumption of a sudden transition from laminar to turbulent conditions,  $\tau_\delta$ ,  $m_\delta$ , and  $q_\delta$  must coincide with the corresponding quantities at the internal limit of the turbulent layer, and, therefore, must satisfy the relations [1]. Hence we can write, eliminating the common factor  $\rho v$ :

$$\frac{u_\delta - u_0}{u_\delta - u_0} (1 - e^{-\eta_\delta}) = \frac{\omega_\delta - \omega_\delta}{\omega_\delta} = \frac{h_{t\delta} - h_{t\delta}}{Q_\delta - H(\omega_\delta)} \dots \dots [11]$$

Observing now that from Equations [6] and [9] and the definition of  $h_t$  we have

$$h_{t\delta} - h_{t\delta} = Q_\delta - Q_\delta - H(\omega_\delta) + H(\omega_\delta)$$

and introducing for brevity

$$\Phi = \frac{u_\delta - u_0}{u_\delta - u_0} (1 - e^{-\eta_\delta}) = \left( \frac{u_\delta - u_0}{u_\delta - u_0} - 1 \right) (1 - e^{-\eta_\delta}) \dots [12]$$

We can rewrite Equations [11] in the following way

$$\frac{\omega_\delta}{\omega_\delta} = \frac{Q_\delta - H(\omega_\delta)}{Q_\delta - H(\omega_\delta)} = 1 + \Phi \dots \dots \dots [13]$$

so that introducing the value of  $Q_\delta$  obtained from Equation [10] and from Equation [13] itself, we obtain

$$Q_0 e^{Pr \cdot \eta_\delta} = \frac{Q_\delta - H(\omega_\delta)}{1 + \Phi} + H \left( \frac{\omega_\delta}{1 + \Phi} \right) + Pr e^{Pr \cdot \eta_\delta} \int_0^{\eta_\delta} H e^{-Pr \cdot \eta_1} d\eta_1 \dots [14]$$

where, in the last integral,  $H$  is a function of  $\eta_1$  through  $\omega$ , given by Equation [8] with  $\omega_\delta = \omega_\delta / (1 + \Phi)$  (Equation [13]).

Equation [14] represents the solution to our problem.

Supposing we know the values of  $(u_g - u_o)/(u_s - u_o)$  and  $\delta$ , and the function  $H(\omega)$ , we can compute  $\eta_s$  for a prescribed  $\rho_o v_o$  from Equation [7] and obtain  $\Phi$  from Equation [12] (see section 7). If the conditions of the gases outside the boundary layer are known, that is, if we know  $h_g$ ,  $\omega_g$  and  $Q_g$  (Equation [9]), all the quantities on the right-hand side of Equation [14] can be evaluated, and we can therefore deduce  $Q_o$ . Vice versa, if  $Q_o$  is known, the value of  $\rho_o v_o$  which satisfies Equation [14] gives the value of the necessary coolant consumption.

If the coolant is inert, we have simply to take  $H = 0$ , and Equation [14] is reduced to

$$\frac{Q_g}{Q_o} = (1 + \Phi)e^{Pr \cdot \eta_s} \dots \dots \dots [15]$$

which corresponds to the developments of Rannie in the case of porous cooling and of Sloop in the case of film cooling.

**6 Simplifications and Particular Cases**

The computation of the integral on the right-hand side of Equation [14] requires the knowledge of the function  $H(\omega)$  in the laminar sublayer, that is, between  $\omega_g$  and  $\omega_s = \omega_g/(1 + \Phi)$ . Now in practical applications,  $1 + \Phi$  is generally between 2 and 4 (as it can be deduced from section 7) so that  $\omega_s$  is considerably smaller than  $\omega_g$ . This means that  $\omega_s$  and, of course, also  $\omega_o$  are generally outside the range of concentrations in which the value of  $H(\omega)$  can be established by thermodynamical computations, since at this low value of the oxygen concentration the products of combustion are not well determined and the temperature is decreased considerably. Since the experimental data also do not give sufficient information, the only thing to do for the time being is to guess a reasonable shape of  $H(\omega)$  for low values of  $\omega$  and perform the applications with this shape. Of course, this procedure will not give certain results, since the final answer is considerably affected by the shape of  $H(\omega)$ ; but at least it will give a way of evaluating the importance of the effects of the reactivity of the coolant. One of the simplest assumptions we can make about  $H(\omega)$  is that  $H$  is proportional to  $\omega$  for small values of  $\omega$ , that is, the thermal effect of every addition of oxygen is proportional to the added amount. This can be considered a sound physical assumption in the case of fast-reacting propellants if the temperature is not so low that the reactions are frozen; and has the advantage of allowing simple expressions. Making use of this assumption, with  $\omega$  given by Equation [8], we find

$$\int_0^{\eta_s} He^{-Pr \cdot \eta_1} d\eta_1 = \frac{1 - e^{-(Sc - Pr)\eta_s}}{Sc - Pr} e^{-Pr \cdot \eta_s} H\left(\frac{\omega_g}{1 + \Phi}\right)$$

If, as we have already observed, the Schmidt number is sufficiently close to the Prandtl number, the numerator of the first fractional factor on the right-hand side can be expanded in powers of  $Sc - Pr$ . Stopping the expansion at the first power the result is the expression

$$\int_0^{\eta_s} He^{-Pr \cdot \eta_1} d\eta_1 \cong \eta_s e^{-Pr \cdot \eta_s} H\left(\frac{\omega_g}{1 + \Phi}\right)$$

independent of the Schmidt number. Replacing this expression into Equation [14], we can put the result under the form

$$\frac{Q_g}{Q_o} = \frac{(1 + \Phi)e^{Pr \cdot \eta_s}}{1 + \frac{H(\omega_g)}{Q_o} \left[ (1 + Pr \cdot \eta_s) \frac{1 + \Phi}{H(\omega_g)} \cdot H\left(\frac{\omega_g}{1 + \Phi}\right) - 1 \right]} \dots [16]$$

We can now specify the form of the left-hand side in the three different kinds of cooling.

*Porous cooling.* The coolant is a gas, and  $r_c$  must be taken zero. Therefore Equation [16] with

$$\frac{Q_g}{Q_o} = \frac{h_g - h_c}{h_o - h_c} \dots \dots \dots [17]$$

affords the relation between the fluid-dynamical conditions and the relative value of the enthalpy excess of the main gases with respect to the enthalpy excess at the wall.

*Film cooling.* As we have already observed, the coolant in the film is heated in an initial stage to a temperature  $T_o$ , which coincides practically with the boiling temperature under the prescribed pressure; after which we can take approximately  $h_c = h_o$ . We obtain, therefore

$$\frac{Q_g}{Q_o} = 1 + \frac{h_g - h_o}{r_o} \dots \dots \dots [18]$$

If the conditions in the main gases and the properties of the coolant are known, this quantity has a definite value, and the only variable left is  $\rho_o v_o$  (entering in the expression of  $\eta_s$ ) which can therefore be determined from Equation [16].

*Sweat cooling.* In this case again  $r_c \neq 0$  and  $T_o$  is practically the boiling temperature, but  $h_o \neq h_c$ ; so that

$$\frac{Q_g}{Q_o} = \frac{h_g - h_c + r_c}{h_o - h_c + r_c} \dots \dots \dots [19]$$

and Equation [16] represents again, as in the case of porous cooling, a relation between the two variables  $\rho_o v_o$  and the enthalpy excess  $h_o - h_c$  at the wall, once the conditions in the main stream and the properties of the coolant are known.

Independently of the simple case when  $H = 0$  which, as we have already observed, gives place to Rannie's Equation [15], the other case of particular interest is the case where the coolant is the fuel itself. In this case the enthalpy in the main stream is determined by the equation

$$h_g - h_c = H(\omega_g) - r_c$$

in the assumption that the conditions of the fuel in front of the injector are the same as those of the coolant before the cooling process. This appears to be a satisfactory assumption for the sweat cooling and the porous cooling; but not as good for the film cooling, since, as already observed, in the initial stages of existence the film receives from the gases a certain amount of heat, which has no correspondence in the fuel to the injector. But, if this amount of heat is small with respect

to the thermal effect of the combustion, the equation can still be assumed to be approximately satisfied; in this case Equation [16] can be simplified because  $Q_g = H(\omega_g)$ , and becomes

$$\frac{Q_g}{Q_0} = \frac{(1 + \Phi)e^{Pr \cdot \eta_s}}{(1 + Pr \cdot \eta_s) \cdot \frac{1 + \Phi}{H(\omega_g)} \cdot H\left(\frac{\omega_g}{1 + \Phi}\right)} \dots \dots \dots [20]$$

Let us now consider the denominator of this expression. The first factor is always larger than unity. The product of the last two factors depends on the shape of the function  $H(\omega)$ . Fig. 1 shows three possible cases: we see, for instance, that if the  $H, \omega$  curve presents always its concavity to the  $\omega$ -axis, then the aforesaid product is always larger than unity (Fig. 1a). The opposite can be true if the curve presents an inflexion point (Figs. 1b and 1c). Now in the range of  $\omega$  in

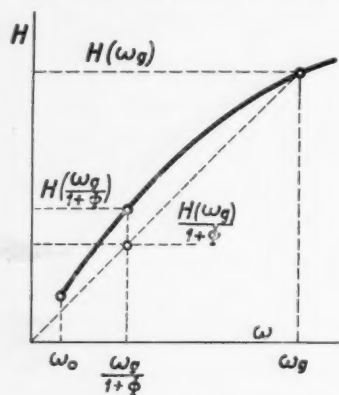


FIG. 1a

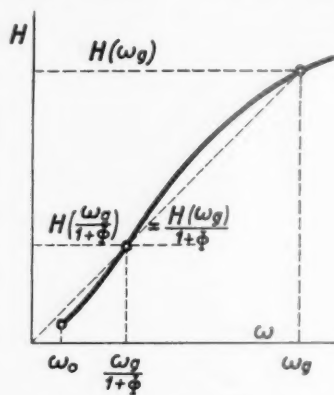


FIG. 1b

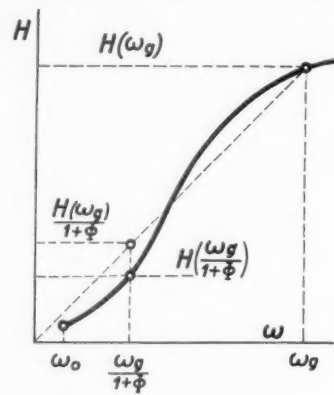


FIG. 1c

which  $H$  can be determined by thermodynamical computation, the first eventuality seems to be the rule; for the remaining values of  $\omega$  we have insufficient quantitative information, but from a qualitative point of view it seems safe to say that even if there is an inflexion point the value of the product is not likely to go much below unity. Therefore the denominator of Equation [20] is generally larger than unity and the comparison with Equation [15], remembering the meaning of  $Q_g/Q_0$  from Equations [17], [18], and [19], shows that the cooling effectiveness is decreased by the reactivity of the coolant.

In particular this is true in the case of Fig. 1b, when Equation (20) takes its simplest form

$$\frac{Q_g}{Q_0} = \frac{(1 + \Phi)e^{Pr \cdot \eta_s}}{1 + Pr \cdot \eta_s} \dots \dots \dots [21]$$

As we have already observed, these results can be considerably affected for different  $H(\omega)$ , and for very slow-reacting propellants the conclusion can be fundamentally changed.

## 7 Numerical Application

We shall consider in this section the numerical results of Equation [21], which has the advantage of containing a minimum number of parameters. In order to be able to perform the numerical computations we require

a way of relating the quantities  $\eta_s$  and  $\Phi$  to the known characteristics of the main flow.

Unfortunately, very little is known today on the characteristics of two-phase boundary layers. A recent experimental investigation (5) shows that in a certain way the annular liquid film surrounding a gaseous stream behaves as a part of a single-phase boundary layer, and that it remains laminar only as far as its thickness is smaller than the thickness of the laminar sublayer in a single-phase boundary layer. If this condition is satisfied, the film is smooth and stable and the transition from laminar to turbulent takes place in the gas stream. If, on the contrary, the film is too thick, transition takes place in the liquid phase and all of the gaseous boundary layer is turbulent. In the second case the results from the computations presented in this paper can only give a lower limit to the liquid

consumption, as mentioned previously. But if the film is sufficiently thin, the theory may be applied after having computed  $\eta_s$  and  $\Phi$  with an approximate method that can be derived from the results of Ref. 5. However here we have used a rougher approach, which is strictly analogous to Rannie's procedure and gives at least a general idea on the behavior of the process. Rannie computes the relations from the ordinary theory of isothermal, incompressible turbulent boundary layer, and then applies the results to the present case. For constant density we can write the three relations

$$\tau_0 = c_F \frac{\rho(u_g - u_0)^2}{2} = \mu \frac{u_g - u_0}{\delta} = \rho u_\tau^2 \dots \dots \dots [22]$$

the first of which defines the friction coefficient  $c_F$  for a relative velocity  $u_g - u_0$  with respect to the liquid boundary ( $u_0 = 0$  if the wall is solid); the second assumes an approximately constant shearing stress throughout the laminar sublayer; and the third defines the characteristic velocity  $u_\tau$ . We obtain first

$$u_\tau = (u_g - u_0) \sqrt{\frac{c_F}{2}} \dots \dots \dots [23]$$

Moreover, as the thickness of the laminar sublayer in the hypothesis of a sharp transition to the turbulent conditions satisfies to

$$\frac{\delta u_\tau \rho}{\mu} = y_*$$



$y_*$  being a constant, we can immediately write, using Equation [23]

$$\frac{\delta(u_\gamma - u_0)\rho}{\mu} = y_* \sqrt{\frac{2}{c_F}} \dots\dots\dots [24]$$

Again from Equation [22] we find, therefore

$$\frac{u_g - u_0}{u_\delta - u_0} = \frac{\mu}{\delta(u_g - u_0)\rho} \cdot \frac{2}{c_F} = \frac{1}{y_*} \sqrt{\frac{2}{c_F}} \dots\dots\dots [25]$$

On the other hand we have from Equation [7]

$$\eta_\delta = \frac{\rho_0 v_0}{\rho g(u_g - u_0)} \cdot \frac{\rho g(u_g - u_0)\delta}{\mu_m} \dots\dots\dots [26]$$

If we indicate with  $w_c$ ,  $w_g$  the mass velocities of the coolant and of the gases, and if  $u_0$  is small with respect to  $u_g$ , the first factor on the right-hand side can be written

$$\frac{\rho_0 v_0}{\rho g(u_g - u_0)} \cong \frac{\rho_0 v_0}{\rho g u_g} = \frac{w_c}{w_g} = \frac{W_c}{W_g} \cdot \frac{A}{S} = \frac{W_c}{W_g} \cdot \frac{d}{4l} \dots\dots [27]$$

the last two steps being derived in the assumption that  $\rho_0 v_0$  is the same on all of the cooled surface  $S$ ;  $W_c$  and  $W_g$  representing, respectively, the total flow of coolant and of propellants, and  $A$  being the section of the cylindrical combustion chamber of diameter  $d$  and length  $l$ . In the last two steps of Equation [27] the cooling of the exhaust nozzle and of the injector head is not considered. The second factor of Equation [26] can be roughly assumed to coincide with Equation [24].

Finally we have

$$\eta_\delta \cong \frac{w_c}{w_g} \cdot y_* \sqrt{\frac{2}{c_F}} \dots\dots\dots [28]$$

so that  $\eta_\delta$  and  $\Phi$  (Equation [12]) can be computed from Equations [25], and [26] once  $y_*$  and  $c_F$  are known. If we take with Rannie<sup>8</sup>

$$y_* = 5.6; \quad c_F = 0.046 \text{ } Re^{-0.2}$$

the Reynolds number being based on the conditions in the main stream and on the diameter of the chamber, we obtain

$$\frac{u_g - u_0}{u_\delta - u_0} = 1.175 \text{ } Re^{0.1}; \quad \eta_\delta = \frac{w_c}{w_g} \cdot 36.8 \text{ } Re^{0.1} \dots\dots [29]$$

The relation between  $Q_g/Q_0$  and  $w_c/w_g$ , as computed using the values of Equation [29] and Equation [12], are represented by the curves of Fig. 2.  $Q_g/Q_0$  is expressed explicitly by Equations [17], [18], and [19] for the three kinds of cooling. The curves are computed for  $Re = 10^5$  and  $Re = 10^6$  since the values for practical rocket motors are generally contained in this range. The comparison between the case of inert coolant from Equation [15] and the special case of reactive coolant represented by Equation [21] shows clearly the increase coolant consumption due to the reactivity of the coolant.<sup>9</sup> At given values of  $Q_g/Q_0$  the increase is of

<sup>8</sup> The case where all of the gaseous layer is turbulent can be obtained from the preceding relations with  $y_* = 0$  and  $\eta_\delta = 0$ . The resulting  $\Phi$  is  $2w_c/c_F w_g$ .

<sup>9</sup> It must be emphasized again that these results correspond to fast-reacting propellants.

the order of 75 to 100%. As a practical example let us consider the film cooling of a chamber with  $h_0 = 1400$  cal/gr (corresponding approximately to the gases of combustion of gasoline and oxygen at maximum trust) and two values of the latent heat:  $r_c = 80$  cal/gr (corresponding to gasoline), and  $r_c = 550$  cal/gr (corresponding to water). We obtain from Fig. 2 the following table of  $w_c/w_g$

TABLE I

Coolant→	$Re = 10^5$		$Re = 10^6$	
	Inert	Reactive	Inert	Reactive
$r_c = 80$	$1.95 \cdot 10^{-2}$	$3.46 \cdot 10^{-2}$	$1.35 \cdot 10^{-2}$	$2.46 \cdot 10^{-2}$
$r_c = 550$	$0.55 \cdot 10^{-2}$	$1.0 \cdot 10^{-2}$	$0.35 \cdot 10^{-2}$	$0.58 \cdot 10^{-2}$

From these values, the values of the actual coolant consumption can be derived from Equation [27] for given values of  $l/d$ . With  $l/d = 1.5$  we have the following table of  $W_c/W_g$

TABLE II

Coolant→	$Re = 10^5$		$Re = 10^6$	
	Inert	Reactive	Inert	Reactive
$r_c = 80$	11.7%	20.7%	8.1%	14.7%
$r_c = 550$	3.2%	6.0%	2.1%	3.5%

For instance, at  $Re = 10^6$ , if we use water as a coolant (inert), we have a minimum coolant consumption of 2.1%, but, if we use the gasoline itself (reactive), the coolant consumption becomes 14.7%. Of the total increase of 12.6%, 6% is due to the decrease of  $r_c$  and 6.6% to the reactivity.

The derivation of Equation [21] has been performed using several drastic assumptions and simplifications. The assumptions of sections 1 and 2 have been introduced in order to obtain the solution in its general form [14], some of them having mainly the purpose or mathematical simplification [assumptions (b), (i)], the others having not only the same purpose but also the one of replacing our very incomplete knowledge of the chemical kinetics of the process [assumption (g), (h)] or of some fluid-dynamical effects [assumption (f)]. The refinement of these assumptions seems to be possible only through considerable mathematical com-

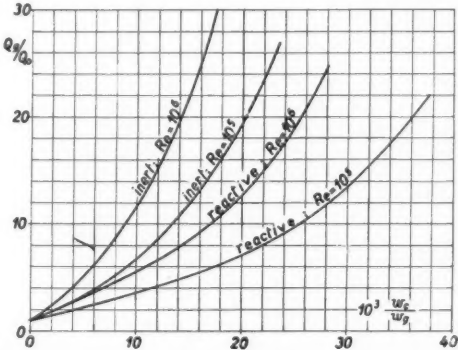


FIG. 2

plications. The rest of the assumptions have been used in the simplification of Equation [14], or in the computation of numerical values. Here, also, the refinement of the very rough evaluations of section 7 would require a much deeper knowledge of the fluid dynamical processes in a turbulent boundary layer and of their dependence upon the thermal conditions. As already mentioned, some help could come from the results of Ref. 5. On the other hand, the approximations introduced at section 6 can easily be avoided when the particular set of propellants and coolant used is specified and the curve of  $H(\omega)$  is more exactly determined. We observe also that the computations can be performed without the simplification of  $Sc$  close to  $Pr$ , and without the restrictive assumption  $Q_0 = H(\omega_0)$ .

Finally we observe that the evaluation of the coolant consumption in the initial stages of the film existence, or in the nozzle, would require the introduction of the  $x$ -derivatives, introduction that cannot be done without considerable mathematical complication.

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### Large-Scale Production and Handling of Liquid Hydrogen

(Continued from page 322)

flowing hydrogen should be grounded to prevent the occurrence of a static spark.

#### 3 General Precautions

(a) Protection of personnel from minor explosions, pressure failures, and other equipment failures may be accomplished by providing metal explosion shields either around the critical sections or by utilizing a panel board which would afford adequate protection.

(b) Precaution against explosions caused by oxygen impurities may be taken by providing continuously recording oxygen analyzers such as those available from Arnold O. Beckman, Inc., of Pasadena, Calif. Suitable contamination alarms may be provided to permit shutting down the plant before hazardous conditions develop.

(c) Careful attention should be given to the hydro-

gen compressors. Preferred units for use are those which have no first-stage-to-atmosphere packing such as the trunk-piston type.

(d) All lubricant in vacuum pumps, compressors, etc., which may come in contact with hydrogen-air mixtures should be noncombustible, such as tricresyl phosphate.

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# Morphological Features of the Isothermal Conversion of Chemical Energy Into Propulsive Energy<sup>1</sup>

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**Editor's Note:** At least seven years ago, F. Zwicky called attention generally to the importance of isothermal expansion processes in the conversion of chemical or thermal energy into jet energy. This idea has been analyzed in various aspects since that date, and recently two papers (4, 5) were submitted independently to the JOURNAL on the subject. In the belief that a formal restatement of the idea would be of interest, the Editor-in-Chief invited the following paper from Prof. Zwicky.

ON VARIOUS previous occasions (2)<sup>2</sup> the author has briefly discussed the morphological features of the conversion into propulsive power of the energy which is released in chemical reactions.

The four basic modes of conversion are represented by two reversible thermomechanical processes and by two reversible thermoelectric modes of generation of electric current and of power.

Of the two ideal limiting thermomechanical processes, the adiabatic conversion of the chemical reaction heat into kinetic energy has been used most widely. On the other hand, the isothermal expansion of the chemical reaction products generated in a combustion chamber has received much less attention. It is desirable to emphasize a few interesting features of this isothermal or (in practice) semi-isothermal conversion.

## A Inadvertent Occurrence of Semi-Isothermal Expansion

One first notices that even a strictly adiabatic expansion which the products of a chemical reaction undergo in a jet motor may, at least in part, proceed by itself and automatically along a more or less constant level of temperature. That is, an adiabatic process with no heat added or subtracted from the outside may, under certain circumstances, assume the appearance of an isothermal process. Specific examples of such circumstances are:

(a) One or more components of an expanding and cooling gas mixture condense when their vapor (or sublimation) pressure  $p_b$  and their boiling point (or sublimation) temperature  $T_b$  is reached. Assuming perfect and instantaneous heat exchange between the remaining gas and the droplets or solid particles formed, the expansion of this remaining gas will be isothermal until

all of the condensable components are liquefied or solidified. The heat necessary for this isothermal expansion is furnished by the heat of evaporation which is liberated by the condensing gases.

(b) Continuing the expansion discussed in (a), it may happen that the total specific heat of the condensed droplets and particles in the jet considered is great compared with the specific heat of the remaining gases. These gases, while expanding, will then be effectively heated by the droplets and the particles, provided that the latter are the smaller in size, the faster is the expansion. As a consequence, the expansion of the gases would appear modified from the strictly isentropic toward a more isothermal process.

(c) Intrinsic heating during the expansion of a gas may occur if gases which are dissociated at very high temperatures recombine. The heat of recombination may then keep the expanding gases at a more or less constant temperature.

## B Artificial Systematic Maintenance of Constant Temperature During Expansion

There is a considerable number of cases of both jet engines and of pumping systems associated with jet engines in which the use of isothermal processes is both of basically theoretical and of practical importance. In this short paper one cannot hope to treat this subject exhaustively. The following three illustrations may suffice to arouse further interest: (1) Avoidance of excessively high temperatures through the use of isothermal processes. (2) Exploitation of available *free enthalpy* of reaction rather than only of enthalpy (heat) of reaction. (3) Increase of efficiency of pumping systems through the use of isothermal processes.

## I Limiting the Combustion Chamber Temperature

For purposes of illustration, all irrelevant features of the processes involved in the chemical reactions and in the conversion of heat into propulsive energy shall be omitted.

It is assumed that per mole of the reagents used, an amount of heat  $H$  is generated. It is further assumed that at all temperatures considered, the average molecular weight of the gaseous reaction products retains a constant value  $\mu$  and that the average specific heats  $C_p$  and  $C_v$  per mole at constant pressure and constant volume are also constant in the whole temperature range.

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<sup>2</sup> Numbers in parentheses refer to the Bibliography on page 342.

For the ratio of the specific heats we write  $\gamma = C_p/C_v$ , and for all quantities per gram,  $h = H/\mu$ ;  $c_p = C_p/\mu$ , etc.

The limiting absolute temperature  $T_L$  which the reaction products may reach as a result of the reaction of our propellants is

$$T_L = T_0 + h/c_p \dots \dots \dots [1]$$

where  $T_0$  is the ambient absolute temperature. The admissible chamber temperature, however, shall be limited to a value  $T_{\max}$  where  $T_{\max} < T_L$ .

In order to achieve this limitation of temperature, several procedures may be followed. Consider ( $\alpha$ ) the use of inert additives to hold the temperature down, and ( $\beta$ ) adiabatic combustion until  $T = T_{\max}$  is reached, followed by subsequent isothermal expansion along a line  $T = T_{\max} = \text{const}$  and pressure  $p$  changing from  $p_c$  to  $p_c'$ , and finally adiabatic expansion from  $p_c'$  to the exhaust pressure  $p_e$ , which is equal to the pressure of the medium surrounding the exhaust of our engine. We now proceed to intercompare the efficiencies of the processes ( $\alpha$ ) and ( $\beta$ ) among themselves as well as relative to the efficiency of the conversion of heat into propulsive energy by the ideal isentropic expansion of the reaction products from  $T_L, p_c$  to  $T_e, p_e$ .

In the following, the general formulas<sup>3</sup> for the various cases are given and, for illustration, they are applied to reaction products for which we have arbitrarily chosen certain numerical values of their thermal properties.

*Thermal Properties Chosen for the Reaction Products.* Mean molecular weight  $\mu = 30$  independent of  $T$ . Heat of reaction available

$$H = 34,400 \text{ cal/mole}$$

or

$$h = 1147 \text{ cal/gram} \dots \dots \dots [2]$$

Ratio of the specific heats  $\gamma = C_p/C_v = 1.3$  independent of  $T$ ; universal gas constant  $R = 1.9872 \text{ cal/mole}$ ; thus  $C_p = 8.6 \text{ cal/mole}$  and  $c_p = 0.287 \text{ cal/gm}$ . It is further assumed that any inert matter added to our propellants has the same specific heats. Furthermore, from Equation [1],  $T_L = 4300 \text{ K}$ .

It is then assumed that the combustion temperature  $T_c$ , because of motor characteristics, must be limited to  $T_{\max} = 3300 \text{ K}$ .

( $\alpha$ ) *Addition of Inert Matter to Depress the Chamber Temperature.* If  $\Delta m$  grams of inert matter are added to one gram of the original reaction products, the resulting  $(1 + \Delta m)$  grams at the temperature  $T_{\max}$  have a total heat content

$$\mathcal{H}_{\text{tot}} = (1 + \Delta m)c_p T_{\max} \dots \dots \dots [3]$$

This must be equal to the heat of reaction  $h$  available per gram of the original reagents plus the total heat content of the  $(1 + \Delta m)$  grams of mixture at  $T_0$ . Thus

$$\mathcal{H}_{\text{tot}} = h + (1 + \Delta m)c_p T_0 \dots \dots \dots [4]$$

and, from Equations [3] and [4]

<sup>3</sup> For some of the numerical calculations and discussions the author is indebted to Dr. J. M. Carter.

$$\Delta m = \frac{h}{c_p [T_{\max} - T_0]} - 1 \dots \dots \dots [5]$$

or numerically for the chosen example,  $\Delta m = 0.332$  grams per gram.

This is the mass of inert matter which must be added per gram of the original propellants in order to depress the chamber temperature from  $T_L$  to the allowable  $T_{\max}$ . The heat available per gram of the mixture on the isentropic expansion from  $T_{\max}$  to  $T = 0$ , i.e., from the chamber pressure  $p_c$  to  $p_e = 0$ , is

$$h' = c_p T_{\max} \dots \dots \dots [6]$$

or, in the example  $h' = 947 \text{ cal/gm}$ . In practice, an expansion from  $p_c$  to a finite exhaust pressure  $p_e$  takes place and the heat  $\Delta h' < h'$  is converted into kinetic energy, where

$$\Delta h' = c_p T_{\max} [1 - (p_e/p_c)^{(\gamma-1)/\gamma}] = 947 [1 - (p_e/p_c)^{0.321}] \text{ cal/gm} \dots \dots \dots [7]$$

The exhaust velocity  $v_e$  and the specific impulse which characterize the efficiency of the motor are proportional to  $\sqrt{\Delta h'}$ . In Table 1  $\Delta h'$  for the process ( $\alpha$ ) is plotted as a function of  $p_e/p_c$  and is compared with the maximum heat  $\Delta h$  convertible by the isentropic expansion from  $T_L$  at the available pressure ratio  $p_c/p_e$ .

( $\beta$ ) *Combination of Isothermal and Isentropic Expansion.* In this process an initial step of the combustion is used which results in the allowable chamber temperature  $T_{\max}$  rather than  $T_L$ . This may be done in practice by injecting at first only a fraction of the available oxidizer. The remaining fraction is then gradually introduced further downstream in such a fashion as to maintain isothermal expansion of the combustion products from the pressure  $p_c$  to  $p_c'$ , which one can determine. (It should be pointed out that this thermodynamic analysis applies only to heat addition during the expansion process and does not account for the mass addition and its possible effects.) From  $p_c'$  to  $p_e$  the gases are again expanded isentropically.

The initial combustion to  $T_{\max}$  consumes per gram of propellants a heat

$$h_1 = c_p [T_{\max} - T_0] = 0.287 \times 3000 = 861 \text{ cal/gm} \dots [8]$$

The total heat of reaction which is available is  $h = 1147 \text{ cal/gm}$ . There remains thus

$$h_2 = h - h_1 = 286 \text{ cal/gm} \dots \dots \dots [9]$$

to be supplied for isothermal expansion from  $p_c$  to  $p_c'$ . This intermediate pressure  $p_c'$  is obtained from the requirement that the isothermal work of expansion per mole  $\mu$  from the pressure  $p_c$  to the intermediate pressure  $p_c'$  is equal to the heat  $\mu h_2$  which is fed in isothermally, thus, per mole

$$\mu h_2 = RT_{\max} \ln p_c/p_c' \dots \dots \dots [10]$$

The determination of  $p_c'$  by the simple formula given in Equation [10] assumes that the effect of mass addition on the pressure distribution can be neglected. For the general argument of this paper, this neglect is justifiable since  $\Delta h''$  is usually not sensitive to the precise value of  $p_c'$ . There is an isentropic expansion from  $p_c'$



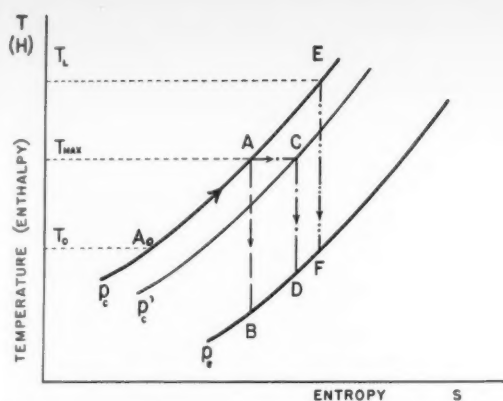


FIG. 1 THREE THERMODYNAMIC PROCESSES FOR THE CONVERSION OF CHEMICAL ENERGY INTO PROPULSIVE ENERGY

$A_0 \rightarrow A \rightarrow B$  = process ( $\alpha$ ) of the text  
 $A_0 \rightarrow A \rightarrow C \rightarrow D$  = process ( $\beta$ ) of the text  
 $A_0 \rightarrow E \rightarrow F$  = optimum adiabatic expansion

to the pressure  $p_e$  of the medium which surrounds the engine, during which process the heat  $\Delta h''$  per gram is converted into kinetic energy. It is

$$\Delta h'' = c_p T_{\max} [1 - (p_e/p_c')^{(\gamma-1)/\gamma}] \dots [11]$$

The total heat  $\Delta h_{\text{comb}}$  which we can convert into kinetic energy by using the described combination of isothermal and of isentropic expansion thus is

$$\Delta h_{\text{comb}} = h_2 + \Delta h'' \dots [12]$$

where  $h_2$  and  $\Delta h''$  are given by Equations [8], [9], and [11].

In Fig. 1 the processes ( $\alpha$ ) and ( $\beta$ ), as well as the ideal expansion from  $p_c$  and  $T_L$  to  $p_e$  and  $T_e$  are represented diagrammatically. The heat per gram converted in the ideal expansion is

$$\Delta h = c_p T_L [1 - (p_e/p_c)^{(\gamma-1)/\gamma}] \dots [13]$$

In Table 1 is listed the efficiency of the three processes described using the numerical values of our assumed example. It is seen that  $\Delta h_{\text{comb}}$ , while only in-

TABLE 1  
(All heats in cal/gm)

$p_c/p_e$	$p_c'/p_e'$	$\Delta h'$	$\Delta h_{\text{comb}}$	$\Delta h$
20	5.38	474	592	617
50	13.45	564	715	..
100	26.90	629	791	..
200	53.80	668	857	..
500	134.50	721	928	..
1000	269.0	755	966	..
2000	538.0	783	1012	..
5000	1345	815	1055	1061

significantly smaller than  $\Delta h$ , is in all cases about 25 per cent greater than  $\Delta h'$ .

If parasite effects were considered such as heat losses, it would be found that  $\Delta h$  presumably suffers greater depletion than  $\Delta h_{\text{comb}}$  such that the combinatory isothermal-isentropic expansion comes out on top.

## II Free Enthalpy Versus Enthalpy

### (Extraordinary Conversions of Heat into Kinetic Energy)

As a consequence of the first and the second law of thermodynamics, the following general result for the conversion of heat into useful mechanical or electrical energy can be derived.

If a material system  $\Sigma$  undergoes a change from an initial state  $\Sigma_i$  to a final state  $\Sigma_f$ , both of which states are in equilibrium under the same external pressure  $p_e$  and temperature  $T_e$ , the maximum amount of useful energy<sup>4</sup> or work  $W$  which can be gained per mole is

$$W = F_{pi} - F_{pf} = \Delta F_p \dots [14]$$

where

$$F_p = E - TS + pV = H - TS \dots [15]$$

Here  $E$  and  $S$  are the internal energy and the entropy per mole, while  $V$  is the molar volume.  $F_p$  is therefore appropriately called the *free enthalpy*.

Although in the operation of most jet engines  $\Delta H = H_i - H_f > 0$ , it must not be assumed that this an absolute requirement. Actually the necessary condition which must be fulfilled by the propellants and their reaction products at constant  $T$  and  $p$  is

$$\Delta F_p > 0 \dots [16]$$

On the other hand, the heat of reaction may be *positive*, *negative*, or *zero*. It will thus be useful to discuss briefly examples of three types of jet engine operations for all of which Equation [16] is satisfied, but for which (1)  $\Delta H > 0$ , (2)  $\Delta H \cong 0$ , and (3)  $\Delta H < 0$ . In all of these cases the use of isothermal expansion and contraction plays a vital role if maximum efficiency is sought.

#### 1 The Normal Case, $\Delta H > 0$

There are three subcases, namely ( $\alpha$ ), ( $\beta$ ), and ( $\gamma$ ).

( $\alpha$ ):  $\Delta F_p > \Delta H$ . This means that more useful energy  $W$  can be extracted than the heat of reaction. If  $W$  is generated as electrical energy through the conduction of the reaction in an electrolytic cell, this cell has a tendency to cool down. In order to conduct the reaction isothermally, a certain amount of heat  $Q$  must therefore flow from the surrounding medium into the cell. This heat  $Q$  added to  $\Delta H$  can be converted into useful work.

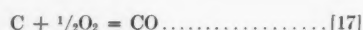
In a thermal engine which operates at *different* levels of temperature, the full amount of useful energy  $\Delta F_p$  can only be generated if a series of large constant temperature reservoirs are available. If only one such reservoir can be used in practice (e.g., the atmosphere or the ocean) the conversion of the maximum available chemical energy  $\Delta F_p$  into either electrical or mechanical

<sup>4</sup> In the ordinary theory of rockets,  $\Delta H$  is always positive and therefore equal to  $H_i - H_f$  rather than to  $H_f - H_i$ , as it is commonly used in thermochemistry. In order to conform with the rocket engineers' use of  $\Delta H$  we likewise assign to  $\Delta F_p$  the positive sign when this amount of free enthalpy is *liberated* during the reaction.

energy can only be achieved through *isothermal* processes; e.g., isothermal expansion during which some heat is extracted from the surrounding medium or reservoir.

In the conventional rocket motors the conversion of the chemical energy into propulsive energy is conducted in two steps, namely, (a) combustion or chemical reaction, with generation of elevated temperatures, and (b) expansion, which, to be reversible, may be isentropic, or isothermal followed by isentropic. Since process (a) is necessarily *irreversible*, the full amount  $\Delta F_p$  of chemical energy available cannot be converted into useful energy by the conventional process (a), (b).

An example for which  $\Delta F_p > \Delta H$  is given by the reaction



for which, at 18 C,  $\Delta H = +26.39$  kcal/gram mole, and  $\Delta F_p = +32.79$  kcal/gram mole.

( $\beta$ ):  $\Delta F_p \cong \Delta H$  is nearly realized in the reaction



for which, at 18 C,  $\Delta H = +94.03$  kcal/gram mole and  $\Delta F_p = +94.24$  kcal/gram mole.

( $\gamma$ ):  $\Delta F_p < \Delta H$ , a subcase expressed by the reaction



for which at 18 C,  $\Delta H = 68.37$  kcal/gram mole, and  $\Delta F_p = +56.69$  kcal/gram mole.

In this subcase it should be pointed out that, although often the heat of reaction  $\Delta H$  is considerably greater than the change in free enthalpy, there is no process which permits converting the full amount of heat into useful mechanical or electrical energy, except if we introduce additional components into the reaction [19]; e.g., inert working gases. In doing so, the subcase ( $\alpha$ ) is again obtained.

## 2 Heat of Reaction, $\Delta H \cong 0$

The simplest example is an ideal gas which, at the absolute temperature  $T_0$  of the surrounding medium, is expanded from the elevated pressure  $p_i$  to the lower pressure  $p_f$ . Since  $E_i = E_f$  and  $p_i V_i = p_f V_f = RT_0$ , we have  $\Delta H = 0$  but  $\Delta F_p$  equal to the heat borrowed from the surrounding medium, which is greater than zero (Equation [10]).

The isothermal expansion of a compressed gas may be used to operate what the author has called an *isothermal hydroduct*. A description of the operation and of the thrust and specific impulse achieved by the isothermal hydroduct was given at the International Congress of Applied Mechanics in Paris in 1946.

## 3 Heat of Reaction Negative, $\Delta H < 0$

The simplest example is perhaps the evaporation of a liquid at the temperature  $T_B$  and the corresponding constant vapor pressure  $p_B$ . It is clearly  $E_f > E_i$ , since

the heat of evaporation has been added to the system. Also  $p_f V_f = p_B V_f > p_B V_i = p_i V_i$ , since the final volume  $V_f$  is assumed to be greater than the initial volume  $V_i$ . Consequently,  $H_f = E_f + p_f V_f > E_i + p_i V_i$ , and  $\Delta H < 0$ . However,  $\Delta F_p > 0$ , since this free enthalpy change is simply equal to  $p_B(V_f - V_i)$ .

A special isothermal hydroduct, for instance, could be operated by injecting an easily evaporable liquid into the water flowing through the duct.

A case which might become of interest in the discussion of interplanetary travel or travel of a satellite around the earth is the use of the sun's radiative energy. Through the use of mirrors or lenses on the satellite, the optimum temperature which can be achieved in the rocket chamber is near to the sun's almost 6000 C. To maneuver a space ship, one only needs short bursts for which one might advantageously use the evaporation of a light element,  $H_2$ , He, or Li in the "sun's furnace." This, then, would constitute an example for Case 3.

## III Isothermal Processes in Pumping Systems

The occasions when one may advantageously use isothermal processes in pumping systems are manifold, and a few qualitative suggestions are given here.

When air or other gases are adiabatically compressed through ram effects in the diffusers of moving aeroducts (ramjets), aeroturbojets, aeroturbo-rockets, aeropulses, etc., these gases are heated. In certain cases it may be useful to increase the density of the compressed gas in the stagnation cross section by carrying off this heat, and to conduct the ram compression more nearly isothermally.

In systems employing liquids or moving through liquids (water), cavitation plays an important role. Cavitation in most cases affects performance adversely, but in some instances it may be used to good purpose; e.g., to properly adjust exit pressures for deep-running hydrojet engines. A survey of methods to either carry away or feed in heat for the purpose of checking cavitation or of enhancing it has never been carried out, but it is herewith suggested as an interesting and useful pastime.

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# Letters to the Editor

This section of the Journal is open to letters not exceeding 600 words in length (or one and one-half columns) devoted to brief research reports or technical discussions of papers previously published. Such letters are published without editorial review, usually within two months of the date of receipt. The style and manner of submission of letters are the same as for regular contributions. (See inside back cover.)

## Thrust and Drag

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### Nomenclature

- $T$  = thrust
- $D$  = external drag
- $T_A$  = available thrust =  $T - D$
- $F$  = stream force =  $pA + mu$
- $A$  = cross-sectional area
- $p$  = static pressure (absolute)
- $m$  = mass flow ( $m_0$  = intake mass flow,  $m_e$  = exhaust mass flow)
- $u$  = velocity
- $f$  = skin friction drag on surfaces exposed to external flow
- $\varphi$  = axial component of the reaction to pressure and skin friction forces exerted on any tangible surface between stations (0) and (1)

Stations (See Figs. 1a and 1b)

- 0 = free stream
- 1 = engine entrance
- e = engine exit

DESPITE the extensive bibliographies that have been built up in reference to the subject of jet propulsion, the practical matter of definition, allocation, calculation, and measurement of thrust and drag components is still so confused that it is seldom possible to compare or correlate performance information between the various types of jet engine discussed by the numerous investigators in this field. In an effort to satisfy, at least partially, what appears to be a real and urgent need for a primer on this subject, the simplified treatment given here may be considered as the Basic English which can be exacted out of the welter of specialized vocabularies that are now incorporated into the language of jet propulsion.

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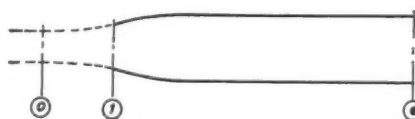


FIG. 1a

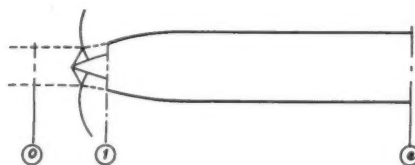


FIG. 1b

DIAGRAMS OF JET ENGINE SHOWING STATIONS

It is not purported that any new concepts are introduced in the following remarks, and, in fact, the content appears trivial at first glance. Nevertheless, a clear and simple formulation is provided as to what one means precisely when the terms thrust, drag, available thrust, additive drag, etc., are used to describe the performance of jet engines. Of course, in addition, such a formulation thus clarifies how one must proceed to calculate or measure these quantities without liability of ambiguousness.

Since there is nothing sacrosanct about the following preliminary codification, it is conceivable that a better way of defining the elements of this rapidly growing technology can be evolved, but it is felt that the following discussion is coherent and consistent and furnishes at least one secure framework within which to study the many problems associated with jet propulsion.

The available thrust of steady-flow jet engines can be calculated as the resultant of either the forces acting on all the tangible internal and external surfaces of the engine, or the forces acting on the air which flows through and around the engine. In the former case, the expression for available thrust can be written in the form

$$T_A = F_e - F_1 - \int_{A_1}^{A_e} p dA - \varphi - f \dots \dots \dots [1]$$

The terms of Equation [1] can all be measured or easily computed, but, with the exception of the last term,  $f$ , they cannot be distinctly identified as thrust or

drag components. In particular, the terms  $\int_{A_1}^{A_2} p dA$  and  $\phi$ , which depend on the meridian contour of the external surfaces, are retained in the equation even in the ideal case of an isentropic external flow. The importance of this drawback can be fully realized when it is considered that the external shape of an engine enclosure is, to a large extent, a matter of compromise between design requirements of the airframe on which the engine is installed, rather than a characteristic property of the engine itself. Therefore, the use of Equation [1] is not convenient when various engines are compared from the standpoint of their characteristic performance.

When the available thrust is calculated as the reaction to the forces that act on the air which flow through and around the engine, the following expression ensues

$$T_A = F_\tau - F_0 - \int_{A_0}^{A_e} p dA - f = m_* u_* - m_0 u_0 + A_e(p_* - p_0) - \int_{A_0}^{A_e} (p - p_0) dA - f \dots [2]$$

In isentropic external flow, the last two terms of Equation [2] vanish. Thus, a grouping of terms suggests itself, whereby

$$D = \int_{A_0}^{A_s} (p - p_0) dA + f \dots \dots \dots [3]$$

hence, remembering that  $T_A = T - D$

$$T = m_s u_s - m_0 u_0 + A_s(p_s - p_0) \dots \dots \dots [4]$$

Equations [3] and [4] are arbitrary but convenient and compatible definitions of thrust and drag. Their compatibility is established by Equation [2]. The definition of the external drag  $D$  as a quantity which vanishes in the absence of irreversible processes in the external flow is in accordance with aerodynamic practice, and further is seen to lead to a definition of thrust which represents a distinct characteristic of the engine, independent of the shape of its enclosure. Thus the thrust  $T$  of any given engine is also the available thrust  $T_a$  which this engine would produce in the ideal condition of external isentropic flow, and as such, it represents the limit of performance which could be approached through aerodynamic refinement of its external surfaces.

In conclusion, the use of Equation [2] is to be recommended in preference to that of Equation [1] in the comparison of potential performance of engines of different types as well as in the correlation of the experimentally measured and theoretically predicted performance of any given engine.

The drag integral  $\int_{A_0}^{A_s} (p - p_0) dA$  may be resolved into the two components

$$(I) \quad \int_{A_1}^{A_e} (p - p_0) dA$$

which is usually referred to as "pressure drag," and

$$(II) \quad \int_{A_0}^{A_1} (p - p_0) dA$$

to which the designation "additive drag" had been applied by some authors (see References 1 and 2 at end of paper).

Then Equation [2] may be written in the form

$$\underbrace{T_A}_{\text{Available thrust}} = \underbrace{m_e u_e - m_0 u_0 + A_e(p_e - p_0)}_{\text{Thrust}} - \underbrace{\int_{A_1}^{A_e} (p - p_0) dA}_{\text{Pressure drag}} - \underbrace{\int_{A_1}^{A_1} (p - p_e) dA}_{\text{Additive drag}} - \underbrace{f}_{\text{Skin friction on external surfaces}} \dots [2']$$

It can easily be verified from comparison of Equations [1] and [2'] that

$$\text{Additive drag} = \varphi + F_1 - F_0 - p_0 (A_1 - A_0)$$

It must be noted that the additive drag is always present when free-stream compression of the working fluid occurs between stations 0 and 1, regardless of whether the external flow is subsonic or supersonic and of whether a central body, such as a spike diffuser, is present or not between these two stations. The separate computation of the additive drag is unnecessary when the *entire* external flow is isentropic, because in such case, as has been noted above, the drag  $D$  is zero (hence additive drag and pressure drag are equal but opposite in sign), i.e.,  $T_A = T$ .

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# Jet Propulsion News

C. F. WARNER, *Purdue University, Associate Editor*

with the assistance of W. G. BOHL

## Rockets and Missiles

**H**OW hot or cold it is from 55 to 75 miles above the earth may be determined by the Army Signal Corps through a new method for gathering facts about the weather, the Department of the Army announced recently. The new method employs an inflated sphere released after an 80-mile ride into space in an Aerobee rocket. The sphere, initially only partly inflated, is carried in a wooden cylinder just behind the nose cone of the rocket. Near the peak of the rocket flight, the four and a half foot nylon sphere, especially developed for these tests, is released. Air from a pressurized cylinder finishes blowing up the sphere and gives it shape. Inclosed in the nylon sphere is an electronic transmitter capable of sending continuous signals to a ground station where the data is recorded. Temperatures in the upper atmosphere can be determined by measuring the rate of fall of the sphere. Neoprene-impregnated, and one-fiftieth of an inch thick, the sphere was made by the Goodyear Tire and Rubber Company of Akron, Ohio.

Meteorologists now believe that at about 55 miles the weather starts warming up from its minus 28 degree Fahrenheit temperature to about 400 degree Fahrenheit at a height of 400 miles. Instrumentation of rockets is part of the Army Signal Corps research and development program. Data to be obtained are expected to help in study of atmospheric changes, long range communications, and design of new operating equipment.

♦ ♦ ♦

The Air Force's X-2 swept-wing, rocket-powered research plane has been delivered by Bell Aircraft Corp. to Edwards Air Force Base, California. The new plane, powered by a Curtiss-Wright Propeller Division 15,000 lb thrust rocket motor, has a K-Monel fuselage and stainless steel wings and tail. It is expected that the X-2 will reach an altitude of over 100,000 ft and a top speed of more than 1500 mph.

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A NORTHROP F-89 Scorpion has been equipped with new type wing-tip tanks which, in addition to carrying an auxiliary fuel supply, will house a number of proximity-fused air-to-air rockets.

♦ ♦ ♦

A ROTARY accelerator built at the Naval Ordnance Laboratory, White Oak, Maryland, can simulate

accelerations in the 100 to 500 g range for several seconds, will be used to study effects of friction and inertia on rocket fuzes. Test items are mounted on a test arm, actuated by compressed air, which reaches full speed in the first 60 degrees of rotation.

♦ ♦ ♦

AEROJET Engineering Corporation has five different liquid propellant rocket engines designed, tested, and ready for quantity manufacture. One of these engines is now in production and others are being flight-tested by possible customers. No additional details of these developments have been released.

♦ ♦ ♦

IT is reported that rocket experiments are being conducted by a group of German scientists near Bremen. To date their tests have been deemed unsatisfactory because of the "poor stability" of the missiles. One of the announced objectives of this work is to develop a rocket vehicle to carry loads such as mail for long distances.

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An SO-6025 French jet fighter made its first flight June 10. The plane is powered by a Nene turbojet and a SEPR rocket motor. No additional information has been released.

## Turbojet Engines

THE Lycoming-Spencer Division of Avco Manufacturing Corporation has been awarded a contract by the Air Materiel Command, U.S. Air Force, for the development of a turboprop aircraft engine. The development program, upon which work is already in progress at the company's Williamsport, Pa., plant, is the first in the gas turbine field to be undertaken by Lycoming, one of the nation's oldest producers of engines for light- and medium-weight aircraft.

The company's turbine development activities are being directed by Dr. Anselm Franz, one of the world's foremost jet engine authorities, who came to this country from Germany following World War II. As head of the engine development division and a director of the Junkers Company in Germany, Dr. Franz designed and developed the famous Jumo 004 jet engine which powered the Messerschmitt ME 262 and was the world's first mass-produced and combat-tested jet engine.

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THE Fiat Company of Modena, Italy, has been

EDITOR'S NOTE: The information reported in this Section has been selected from approved news releases originating with the Department of Defense, private manufacturers, universities, etc., and from published news accounts in journals and newspapers. The reports are considered generally reliable, although no attempt has been made to verify them in detail.

licensed to manufacture spare engine parts for Allison J35-A-29 engines in Republic F-84G Thunderjets sold to USAF and NATO countries. The license agreement was negotiated by representatives of the U. S. Air Force, Fiat, and Allison. Fiat is a manufacturer of Italian motor cars and various other types of equipment. Approximately 70,000 people are employed by the company.

♦ ♦ ♦

PERHAPS the outstanding engine of 1952 is the British high compression ratio Olympus. The basic arrangement of the turbojet's two axial-flow compressors and turbines are shown in Fig. 1. The inner

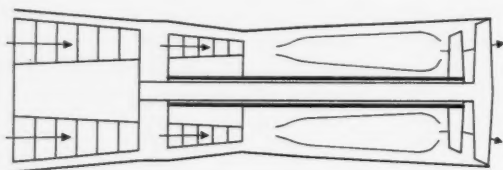


FIG. 1. SCHEMATIC DIAGRAM OF THE OLYMPUS

turbine and compressor are connected by means of a hollow shaft through which runs the independent shaft connecting the outer turbine and compressor. Although this arrangement requires more shafting, bearings, and weight than is used in one-compressor engines, the specific weight of the Olympus is 0.36 lb/lb thrust.

The use of the double compressor makes it possible for the Olympus to achieve a high pressure ratio without the dangers of surging. The surging condition results from the stalling of rear rows of compressor blades. This condition is difficult to prevent when a large number of rows of fixed angle blades are turned at the same speed. The split compressor arrangement overcomes this difficulty by allowing the rear rows of blades, the second compressor, to run faster than the starting rows of blades.

The Olympus exhibits extreme flexibility in control. The throttle may be opened and closed rapidly, increasing the thrust from idling to full power in 3 seconds or less. This performance is achieved without the use of blow-off valves, or "electricrery."

The Olympus is reported to have a diameter of 40 in., and a length of 10 ft without afterburner. The turbojet weighs 3520 lb and develops 9750-lb thrust with a specific fuel consumption of 0.766 lb/lb thrust hr.

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SOCIÉTÉ Rateau of France is developing a turbojet engine, designated the Rateau SRA 301 Berry, that is expected to develop approximately 19,800-lb static thrust.

## Aircraft

DOUGLAS Aircraft Company's Long Beach Division will engineer and build the RB-66 jet reconnaissance bomber for the U. S. Air Force, Donald W. Douglas, Jr. vice-president, military sales, announced. The RB-66

is a twin-jet, swept-wing bomber developed for use of the Tactical Air Command. Based on the A3D, originally designed for the U. S. Navy by the Douglas El Segundo Division, the RB-66 will have a three-man crew. It will be in the 600-700-mph class.

♦ ♦ ♦

THE Navy's new Douglas F4D Skyray, see Fig. 2, is said to be one of the hottest interceptor aircraft in the world today. Thin planform or delta-type wing plus a huge advanced type of jet engine permit the F4D to climb rapidly to high altitudes. The fighter gained its name from its resemblance to the ocean-dwelling manta ray. Now undergoing flight testing

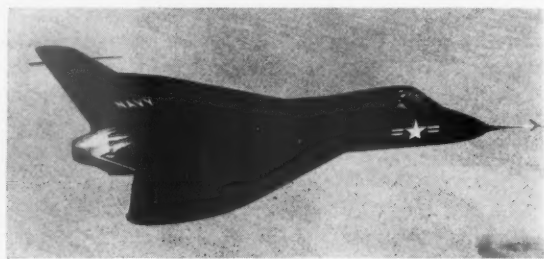


FIG. 2. DOUGLAS SKYRAY, FAST JET-PROPELLED INTERCEPTOR

for the Navy, the Skyray is scheduled for production at the Douglas El Segundo Division.

♦ ♦ ♦

IT is reported that Douglas is ready to build a four-motored commercial jet transport by 1958. At the present time the company has a full-scale fuselage mockup for exhibition to prospective buyers. The plane will have a span of 127 ft and a fuselage length of 134 ft. It is expected that the jet liner will have a domestic range of 2500 mi at 560 mph, and will have a gross weight of 220,000 lb.

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WILLIAM M. ALLEN, President of Boeing Airplane Company, has announced that the Boeing Company has for some time been engaged in a company-financed project which will enable it to demonstrate a prototype jet airplane of new design to the Armed Services and the commercial airlines in the summer of 1954.

♦ ♦ ♦

THE swept-wing XFJ-2 Fury has successfully completed its first carrier suitability tests aboard the *USS Midway* off the Atlantic coast, the Navy has announced. A fast-climbing, better than 650 mile-an-hour jet, the Fury is in the early stages of production at North American Aviation's Columbus, Ohio plant.

The Fury is an advanced version of the FJ-1, the Navy's first operational jet. Completely redesigned, however, the new airplane incorporates many improvements which have resulted from aerial fighting in Korea. It is powered by a General Electric J-47 GE-2 turbojet which is rated at over 5800-lb thrust. This is the same engine powering North American Sabre Jets which have racked up an 8-to-1 kill ratio over Russian-built MIG's in Korea.

The Fury is armed with four 20-mm cannons and has a service ceiling above 45,000 feet and a range of about 1000 miles. Demonstrating the increased complexity of modern aircraft, the new jet required more than 370,000 engineering hours to make its first flight. It incorporates an especially designed wing with a 35-degree sweepback and a hydraulically powered "Flying Tail."

### New Facilities

THE Phillips Petroleum Company has been designated by the U. S. Air Force as the operator of the Bluebonnet Plant, McGregor, Texas, for the production of solid propellant JATO units. According to the USAF, Phillips was selected after detailed consideration of more than 30 firms. Phillips has created a Rocket Fuels Division staffed by key administrative and technical employees of the company, and by trained solid propellant experts. In addition to this division, which will conduct the full-scale development and manufacturing program, the efforts of a large technical group in the company's research division is devoted to the basic development of propellants using petroleum products which have extremely large potential availability.

Although initial research operations are being conducted in Bartlesville, all but the research work will soon be transferred to the plant at McGregor, Texas. This facility, formerly known as the Bluebonnet Ordnance Plant, was built by the Government during World War II and operated as a bomb-loading plant until 1945. The use of this plant will mean a substantial saving in time and money in establishing production of the urgently required JATO units, since there are many facilities in the plant which can be readily adapted to the program.

E. F. Kindsvater, who has been named the manager of the new project, formerly headed the engineering and chemical engineering departments of Phillips Petroleum Company. Emil A. Malick, who will be assistant manager, previously served Phillips as technical consultant and as supervisor of jet development activities.

♦ ♦ ♦

CONSTRUCTION of facilities related to the eventual development of nuclear propulsion for aircraft has been authorized by the U. S. Atomic Energy Commission. Testing facilities will be built at the AEC's National Reactor Testing Station in Idaho. The new facilities are estimated to cost approximately \$33,000,000 and construction is scheduled to begin this summer, it was announced by AEC General Manager M. W. Boyer. During the peak of construction, about 1000 workers will be employed and about 200 employees will be required for operations.

Preliminary design studies for the AEC ground test station have been performed by the Parsons-Macco-Kiewit Company of Los Angeles, Calif., under a subcontract with General Electric. The design, de-

velopment, and fabrication of the prototype aircraft propulsion reactor are being carried out by the Aircraft Gas Turbine Department of General Electric at Lockland, Ohio. The nuclear phases of the project parallel the associated nonnuclear propulsion devices being developed by General Electric under an Air Force contract.

♦ ♦ ♦

PRATT & Whitney Aircraft, already deeply engaged in turbojet engine production and development, will further extend its range in the jet field by developing and building ramjet engines for high-speed guided missiles. The company's entry into ramjet development rounds out completely Pratt & Whitney Aircraft's program in building all types of air-burning engines. The ramjet project, undertaken for the United States Navy, will involve development of various types of ramjet engines.

Pratt & Whitney Aircraft will have the advantage of six years of research on ramjets which has already been done at East Hartford by the Research Department of United Aircraft Corporation. United Aircraft undertook, under the Navy's auspices, a ramjet engine research program in 1946 and has been working on it ever since. A new laboratory, built exclusively for study of ramjet burners, has been operated by United Aircraft's research department since April 6, 1950. The facility is located at the east side of the department's wind tunnel at Rentschler Airport.

The ramjet laboratory, or test stand, a joint project of the Navy's Bureau of Aeronautics and United Aircraft, will continue on this work as part of Pratt & Whitney Aircraft's ramjet program. Many of the people who have been engaged on the research department project will be transferred to P & WA's organization. In addition, P & WA's Andrew Willgoos Turbine Laboratory in East Hartford will be expanded within the next few months to accommodate the ramjet program.

### New Equipment

THE Maxim Silencer Company of Hartford, Connecticut, a pioneer in the development of acoustical devices for the suppression of noise, and a producer of silencers for the first turbojet engines developed in the United States, has developed a silencer which it claims can satisfactorily attenuate noise from a jet engine installed in an aircraft. This silencer, Model VP, is already in use at the Lockheed Aircraft Plant in Burbank, Calif.

Turbojet engines, especially those with afterburners, operate at a noise level that is extremely dangerous to anyone working in the immediate vicinity of their exhaust. Until recently, it was impossible to gage this noise accurately because there were no instruments capable of recording the intensities produced. Such equipment is now available, however, and it seems quite well established that a turbojet with afterburner produces a noise level of approximately 165



decibels, and that the same engine without afterburner is likely to have a noise level of 135 decibels. While a human being wearing ear plugs or ear flaps might work in the vicinity of a noise level of 135 decibels for a short period of time, exposure to a noise level of 150 decibels or above for even a short time may cause permanent injury and even death. Both levels are far above the 85-decibel level at which the Medical Corps of the United States Army says a worker can operate continuously without permanent injury. Maxim says that by simply increasing the length of the silencer it "is able to provide as little or as much noise attenuation as is necessary for the installation, depending upon its location with respect to private homes," etc.

The Maxim VP silencer is cylindrical in shape with an unobstructed cylindrical central passage which permits relatively high flow velocity with a minimum of back pressure. This central passage consists of two concentric tubes of perforated carbon steel, stainless steel, or Inconel running the entire length of the silencer. Between these tubes, glass wool, nickel wool, or copper wool is packed to absorb high-frequency noises. The space between the outer perforated tube and the outside shell of the silencer is divided by headers into chambers acoustically designed to attenuate low-frequency noises. This is the only unit available which offers both high- and low-frequency attenuation. While high-frequency attenuation is readily obtained through the use of splinter panels and other devices, low-frequency noise, which is extremely severe in a turbojet engine, can be controlled only by silencers employing resonating chambers.

The problem of silencing jet engines is complicated by the necessity of reducing the high temperatures they generate without increasing the velocity through the silencer. Exhaust temperatures, as high as 1500 F without an afterburner, and around 3000 F with an afterburner, can be reduced to 450 F, thus permitting the use of relatively inexpensive materials, by the use of a water spray or by the addition of secondary air. However, since the addition of secondary air would increase the volume to be handled, necessitating the use of a large silencer if flow velocity is to be kept below 400 fps (greater velocities create new noises), Maxim recommends the use of a water spray in order to permit the use of a smaller silencer. However, the addition of some secondary air is both desirable, to permit the evaporation of the water spray, and unavoidable. It is not possible to connect the engine perfectly to the approach pipe and water spray leading to the silencer, but North American Aviation has determined that jet engines will operate satisfactorily with secondary air held to a quantity equal to the discharge of the engine.

Besides those mentioned, Maxim has units operating in test stands developed for the Navy Department, the National Advisory Committee for Aeronautics, and in the plants of many manufacturers of engines. It is also furnishing all the silencers for the experimental test stands and the production test stands at the

Wright Aeronautical Corporation in Wood-Ridge, N. J., where the British Sapphire J-65 engines will be built.

\* \* \*

AERODYNAMIC research on supersonic aircraft and guided missiles is being aided by a General Electric dewpoint recorder installed by the National Advisory Committee for Aeronautics in one of the nation's largest supersonic wind tunnels at Langley Field, Va. The G-E instrument, which automatically and continuously measures and records dewpoint temperatures, helps to guard against the condensation of moisture which causes flow disturbances in the test section of the 4 X 4-ft supersonic pressure tunnel.

In attaining velocities up to twice the speed of sound, the air in the test section expands rapidly. This causes a sharp drop in temperature to values of -100 F or lower, depending on operating conditions. Moisture condensation in the test section tends to go up in proportion to the drop in temperature. To guard against this, the G-E recorder is used to monitor the tunnel air to determine the rate at which dry air must be bled into the tunnel to keep the moisture content low enough to avoid excessive condensation.

The complete dewpoint recorder at Langley Aeronautical Laboratory is enclosed in a single floor-mounted steel cabinet. It consists basically of a temperature recorder, a gas chamber, an electronic unit, a two-stage refrigeration system, controls, a chamber light source, and phototubes. Air, pulled in from the wind tunnel, goes through the gas chamber and over a refrigerated mirror which is located in a phototube system. Moisture in the air forms a dew spot on the mirror and a resulting decrease in reflected light which is measured by the phototubes. Signaled by the phototubes, the refrigeration is regulated to maintain the mirror at a temperature which will permit a dew spot to form but not to grow. This temperature, the dewpoint temperature, is transmitted to the recording unit by a thermocouple mounted directly on the mirror.

### New Student Rocket Society Is Formed

The Student Rocket Society, a nonprofit scientific and educational organization, has recently been organized on a national scale by Alfred D. Goldenberg and a group of interested students. ARS member, Z. A. Thypaldos, assisted in the formation of the Student Rocket Society and is now an Adviser of the Society. The purpose of the SRS is to encourage in high school and college students an interest in the fields related to rocketry and astronautics. Mr. Goldenberg believes that the great number of engineers and scientists needed in the jet propulsion industry cannot be supplied by the colleges and universities unless today's students become interested in the field. At the present time, the Student Rocket Society has sections in California, New York, Ohio, Illinois, and Maryland. Mr. Goldenberg, of 1673 Via Solida, Palm Springs, Calif., is the Society's National President.



# American Rocket Society News

H. K. WILGUS, *Associate Editor*

## Full 3-Day Program Scheduled for ARS Seventh Annual Convention, December 3-5, at McAlpin Hotel, New York, N. Y.

THE American Rocket Society will hold its seventh Annual Convention on December 3-5, 1952, at the McAlpin Hotel, New York, N. Y., in conjunction with the ASME Annual Meeting which is being held concurrently at the nearby Hotel Statler, New York, N. Y. The fund of new information scheduled for discussion at the five technical sessions—at which 20 papers will be presented—is an indication of how well the Society reflects the nation's progress in rocket and jet propulsion.

Registration will begin on Wednesday, December 3, at 9:00 a.m. Those attending technical sessions must register at the Registration Desk at the McAlpin Hotel, Broadway at 34th St., New York, N. Y. There is no fee for members of the ARS or of the ASME. For all others a fee of \$5 will be charged for admission to one or all of the technical sessions. Payment of registration fee is not required for attendance at luncheons, dinners, and inspection trips.

The Annual Business Meeting will occur at 10:00 a.m., Wednesday, followed at 12:15 p.m. by the Section Luncheon (open to representatives and officers of the Society).

The technical program follows:

### Session I

#### 2:30 p.m., Wednesday, December 3:

*Chairman:* E. L. Hull, General Electric Company, Schenectady, N. Y.

*Vice-Chairman:* R. J. Thompson, The M. W. Kellogg Company, Jersey City, N. J. A Survey of Combustion Instability in Liquid Propellant Rocket Engines, by R. S. Levine and R. B. Lawhead, North American Aviation, Inc., Aerophysics Laboratory, Downey, Calif.

A Simplified Combustion Analysis System, by R. Neumann, D. Dembrow, W. Berl, and R. Prescott, Applied Physics Laboratory, The Johns Hopkins University, Silver Spring, Md.

Combustion Studies in Rocket Motors, by K. Berman and S. H. Cheney, General Electric Company, Malta Test Station, Schenectady, N. Y.

### Session II

#### 9:30 a.m., Thursday, December 4:

*Chairman:* F. C. Durant III, Vice-President, American Rocket Society.

*Vice-Chairman:* A. G. Haley, General Counsel, American Rocket Society.

The Atmospheres of Earth and Mars in the Light of Recent Physiological Concepts, by H. Strughold, U.S.A.F. School of Aviation Medicine, Randolph AF Base, Texas.

Escape and Survival in the Border Zone of Space, by F. Haber, U.S.A.F. School of Aviation Medicine, Randolph AF Base, Texas.

Establishment of Large Satellites by Means of Small Orbital Carriers, by K. A. Ehricke, Redstone Arsenal, Huntsville, Ala.

Rockets Behind the Iron Curtain, by G. P. Sutton, North American Aviation, Inc., Downey, Calif.

### Session III

#### 2:30 p.m., Thursday, December 4:

*Chairman:* C. F. Warner, Purdue University, Lafayette, Ind.

*Vice-Chairman:* R. L. Duncan, Power Branch, Office of Naval Research, Washington, D. C.

Application of the Ramjet to Aircraft Propulsion, by M. S. Harned, Marquardt Aircraft Company, Van Nuys, Calif.

Application of Analogue Techniques to Control Design for Aircraft Engines, by W. C. Schaffer, Wright-Aeronautical Corp., Wood Ridge, N. J.

Determination of Pressure Time Curve for Motors of Gun-Launched Rockets, by E. F. Lype, Armour Research Foundation, Chicago, Ill.

Telemetry Instrumentation for Rocket Flight Tests, by R. P. Haviland, General Electric Company, Schenectady, N. Y.

#### 7:30 p.m., Thursday, December 4:

##### HONORS NIGHT DINNER

*Speaker:* Lt. Gen. Laurence C. Craigie, U.S.A.F., Deputy Chief of Staff for Development.

*Subject:* On the Practical Translation of Rocket Power to Air Power.

##### Awards:

R. H. Goddard Memorial Award to R. W. Porter, General Electric Company.

C. N. Hickman Award to A. L. Antonio, Aerojet Engineering Corp.

G. Edward Prenday Award to M. J. Zucrow, Purdue University.

Student Award to Richard W. Foster, Purdue University.

### Session IV

#### 9:30 a.m., Friday, December 5:

*Chairman:* Martin Schilling, Redstone Arsenal, Huntsville, Ala.

*Vice-Chairman:* R. B. Foster, Bell Aircraft Corp., Buffalo, N. Y.

The Evaluation of Competing Rocket Power Plant Components for Two-Stage Long-Range Vehicles, by A. L. Feldman, Consolidated Vultee Aircraft Corp., San Diego, Calif.

Design of Liquid Propellant Booster Rockets, by Powel Brown, M. W. Kellogg Co., Jersey City, N. J.

The Effect of Variation of Propellant Density on Rocket Performance, by J. Lorell and A. R. Hibbs, Jet Propulsion Laboratory, California Institute of Technology, Pasadena, Calif.

Hydrogen Peroxide, Problems and Operating Procedures, by G. N. Meckert, Air Force Flight Test Center, Edwards Air Force Base, Edwards, Calif.

Effect of Pump Performance on Liquid Propellant Rocket Design, by J. E. Zimmerman, Rocket Section, Non-Rotating Engine Branch, Wright Air Development Center, Wright-Patterson Air Force Base, Dayton, Ohio.

#### 12:15 p.m., Friday, December 5:

##### AMERICAN ROCKET SOCIETY LUNCHEON

*Speaker:* W. E. Zisch, Aerojet Engineering Corp., Azusa, Calif.; *Subject:* Future Engineering Needs of the Rocket Industry. *Speaker:* Wernher von Braun, Redstone Arsenal, Huntsville, Ala.; *Subject:* Space Superiority, A New Concept for Preservation of World Peace.

### Session V

#### 2:30 p.m., Friday, December 5:

*Chairman:* R. C. Stiff, Aerojet Engineering Corp., Azusa, Calif.

*Vice-Chairman:* R. Youngquist, Reaction Motors, Inc., Rockaway, N. J.

Ignition of Fuel With Nitric Acid, by K. C. Halliday, Eclipse Pioneer Div., Bendix Aviation Corp., Teterboro, N. J.

The Effect of Chemical Reactions Upon Predicted Performance of Rocket Motors, by R. F. Potter, Bell Aircraft Corp., Buffalo, N. Y.

The Nitric Acid Ammonia Propellant Combination for Rockets, by R. J. Thompson, The M. W. Kellogg Company, Jersey City, N. J.

A Modified Sodium Line Reversal Technique for Measurement of Combustion Temperature in Rocket Engines, by M. F. Heidmann and R. J. Priem, National Advisory Committee for Aeronautics, Lewis Flight Propulsion Laboratory, Cleveland, Ohio.

## Third IAF Congress in Stuttgart Attracts 200 Scientists

### ARS Participates

AMERICAN Rocket Society delegates were among the 200 scientists who participated in the Third International Astronautics Federation Congress held at Stuttgart, Germany, Sept. 1-6, 1952. Representatives of rocket societies from Great Britain, Germany, Switzerland, Italy, Sweden, Argentina, Norway, Denmark, Holland, and the United States were present. Societies were reported to be forming in Japan, Australia, and Egypt.

Eugen Saenger, IAF President, presided over the six-day program, and was assisted by Vice-Presidents Andrew G. Haley (American Affairs), ARS General Counsel, and Guenter Loeser (European Affairs); and Josef A. Stemmer, Secretary. F. C. Durant III, ARS Vice-President, represented the Society's interests at the Congress.

IAF affairs occupied the first two days of the Congress. A constitution stating the aims and By-Laws of the Federation was reviewed and adopted unanimously. The American Rocket Society was elected to the Council as voting member for the United States. The Pacific Rocket Society and the Reaction Research Society, which had been represented at the 1951 Congress in London, were elected members, as was the Chicago Rocket Society. Plans were made for study of the problem of a standard method of library classification for the subject of space travel. A program was initiated to standardize nomenclature and symbols among member societies.

Officers of the IAF for 1953 were elected as follows: *President:* Eugen Saenger; *First Vice-President:* Andrew G. Haley, ARS General Counsel; *Second Vice-*

*President:* Leslie R. Shepherd, technical director, British Interplanetary Society; and *Secretary:* Josef A. Stemmer, President of the Swiss Astronautical Society. Special recognition was accorded to Guenter Loeser for his outstanding work in handling business matters of the Federation during the past year. Marvin Hobbs, treasurer of the ARS Washington-Baltimore Section, and Miss Franki L. van der Wal (ARS Southern California Section) were guests for these sessions.

### ARS Papers Presented

The next two and one-half days were devoted to the presentation of technical papers. The ARS contributions to these sessions were: "On Optimizing the Component Proportions of High Performance Rockets," by R. A. Cornog and F. L. van der Wal (presented by Miss van der Wal); "Exposure Hazard from Cosmic Radiation at Extreme Altitude and in Free Space," by Hermann J. Schaefer, U. S. Naval School of Aviation Medicine, Pensacola, Fla.; and "Establishment of Large Satellites by Means of Small Orbital Carriers," by Krafft A. Ehricke, Redstone Arsenal, Huntsville, Ala. At a special open meeting, F. C. Durant III presented a paper for Wernher von Braun, entitled "Space Travel—An International Task."

The Congress closed with a banquet at which the Oberth Medal of the German Society was awarded to Dr. von Braun. The presentation was accepted by Cmdr. Durant in Dr. von Braun's absence. Throughout the entire proceedings of the Congress a strong feeling of good will was evident among the delegates. Language barriers seldom existed and the international aspect of the meetings was most stimulating.

The IAF plans to hold its next Congress on Aug. 10-15, 1953, in Zurich, Switzerland.

## Successful ARS Fall Meeting Held in Chicago

THE most successful Fall meeting of the American Rocket Society was held at the Hotel Sheraton, in Chicago, Ill., on September 9 and 10. The meeting was held in conjunction with the Centennial of Engineering. Three sessions were held and all were fully attended.

At the first session held on the morning of September 9 the following papers were presented: "Upper Atmosphere Research" by Dr. Homer Newell; "The High Altitude Sounding Rocket" by Milton Rosen and R. B. Snodgrass; and "Exposure Hazards from Cosmic Radiation at Extreme Altitude and in Free Space" by Dr. H. J. Schaefer.

Spirited discussion from an audience of over 120 closed the session.

The second session held in the afternoon of September 9 was devoted to problems associated with space travel. "The Satellite Vehicle" was discussed by Dr. Wernher von Braun, and Dr. Heinz Haber presented some of the problems that may be encountered in "Manned Flight at the Borders of Space."

The overflow attendance, exceeding 250 people, completely filled the lecture room, with several listeners standing through the session. Both Dr. von Braun and Dr. Haber answered numerous questions from the audience.

The third session, held on the morning of September 10, was devoted to technical aspects of air-burning jet propulsion engines and rocket engines. The session was well attended, over 120 people being present, and the discussion period was quite active. The following papers were presented: "Problems of Development of Ram Jets for Supersonic Application," by J. O. Charshafian; "The Pulse Jet as a Present Day Power Plant," by Brooks Morris; "On Stability and Control of Liquid Propellant Rocket System," by Y. C. Lee; and "Experimental Investigation of Performance of WFNA-JP-3 Rocket Motors at Different Combustion Pressures," by Dr. M. J. Zucrow and C. M. Beighley.

Much favorable publicity was given to the papers presented at the Tuesday session by the Chicago newspapers.

## New Mexico-West Texas Section Elects Board of Directors

FOLLOWING up the election of 1952 officers at their previous May meeting, the New Mexico-West Texas Section held a business meeting on July 31, 1952, to select a Board of Directors and appoint several committees.

The following members were elected to serve on the New Mexico-West Texas

### PRINCIPAL DELEGATES PRESENT AT THIRD IAF CONGRESS

Argentina	Sociedad Argentina Interplanetaria	Prof. T. M. Tabanera Mrs. H. Balado
Austria	Osterreichische Gesellschaft für Weltraumforschung	Prof. Dr. Friedrich Hecht
Denmark	Dansk Selskab for Rumfarts-Forskning	E. Buch Andersen
Germany	Gesellschaft für Weltraumforschung	Prof. Dr. Karl Schutte Dr. jur H. Gerlach Prof. Dr. H. Konig Hans K. Kaiser Dr. ing J. M. J. Kooy H. F. Michielsen Prof. Antonio Eula Glauco Partel Johan Nicoll Ing. Ake Hjertstrand Ing. Josef A. Stemmer Dr. Rudolf Brunner Arthur C. Clarke Dr. Leslie R. Shepherd John Humphries L. J. Carter F. C. Durant III A. G. Haley (By proxy) H. J. Kaeppler H. J. Kaeppler H. J. Kaeppler
Holland	Nederlandse Vereniging voor Ruimtevaart	
Italy	Associazioni Italiana Razzi	
Norway	Norsk Interplanetarisk Selskap	
Sweden	Svensk Interplanetarisk Selskap	
Switzerland	Schweizerische Astronautische Arbeitsgemeinschaft	
United Kingdom	British Interplanetary Society	
U. S. A.	American Rocket Society  Pacific Rocket Society Reaction Research Society Detroit Rocket Society	

\* Now amalgamated with the larger Stuttgart Society.

Section Board of Directors: Brig-Gen. G. G. Eddy, R. F. Fearn, D. M. Cottler, L. W. Gardenhire, J. S. Piech, L. D. White, and H. L. Karsch, former Section president.

Publicity, Membership, and Program Planning Committees were also appointed and organized.

#### **Tombaugh Speaks at Open Meeting**

The New Mexico-West Texas Section had an open meeting on Oct. 30, 1952, at the Brannigan Library Auditorium, Las Cruces, N. Mex., at which a large audience attended to hear Clyde W. Tombaugh speak on "Is There Life on the Other Planets?"

Mr. Tombaugh, who is with the Flight Determination Laboratory at White Sands Proving Ground, was formerly a staff member at the Lowell Observatory, and while there he discovered the planet Pluto.

A full report of this meeting will be included in the next issue of the JOURNAL.

#### **P. R. Bassett Addresses New York Section Meeting**

P. R. BASSETT, president of the Sperry Gyroscope Company, was the guest speaker at the monthly meeting of the ARS New York Section held on Oct. 17, 1952, in the Engineering Societies Building, New York, N. Y. More than 150 members and guests gathered to hear Mr. Bassett give a talk on "Aircraft and Missile Guidance and Control." Michael Samek, of the M. W. Kellogg Company, Jersey City, N. J., presided.

The main part of Mr. Bassett's talk was taken up with an account of how servo systems have developed in the special field of controlling vehicles in the air. The history of quick-reacting servos, he pointed out, is intimately tied up with the gyroscope. When Mr. Sperry in 1910 built his first gyrocompass, one of the important elements of its success was a servo system than called the "follow-up system." Another early application of the gyroscope, Mr. Bassett stated, was its function as a stabilizer for airplanes.

When World War I broke out in 1914, military flying quickly degenerated from the expected straight and level courses for reconnoitering to dog fighting and extreme maneuvering, and the stabilizer went into a secret category, Mr. Bassett said. The story is briefly that the Navy gave to Sperry in 1915 what was probably the first contract ever placed for a guided missile, he said. During those war years this development was perfected to the point where the missile could be launched by catapult. After World War I, for about ten years, automatic control of airplanes was relegated to doing odd jobs, such as flying small airplanes for artillery practice. But when multiengineed airplanes started to carry passengers and fly long distances, the need

of the automatic pilot came in to relieve the human pilot from fatigue.

In the 1930's electronics entered the field of servo systems and quickly spread into gun controls, searchlight controls, and, finally in 1940, to airplane automatic pilots—a revolution in servomechanisms. However, Mr. Bassett cautioned the audience, servo systems and electronics should never be considered synonymous.

The speaker concluded by pointing out that servos are now so competent that they are no longer assistants to the human. So we now have, he stated, in rockets and guided missiles a remarkable reversal in the realm of pioneering where the servomechanism goes out on its own and reports back to the man on the ground.

#### **Rockets, Past and Future, Discussed at Washington-Baltimore Section Meeting**

OVER 200 persons were present at the September meeting of the Washington-Baltimore Section of the American Rocket Society held on September 26 at the Department of Commerce Auditorium in Washington, D. C. The meeting was announced as a local observance of the Centennial of Engineering and was open to the general public.

Two papers were presented at the meet-

ing. Joseph W. Siry of the Rocket Sonde Research Branch, Naval Research Laboratory, spoke on "Twentieth Century Rocket Research." Fred C. Durant III, Vice-President of the American Rocket Society, spoke on "Rockets and Space Flight."

Mr. Siry traced the history of rockets from their early beginnings in the fire arrow to the present giants such as the V-2 and the Viking. Through the use of slides, Mr. Siry showed how the great accomplishments of the twentieth century stemmed from the work of Goddard and of the American Rocket Society in this country, and from the work of Oberth and the Verein für Raumschiffahrt in Germany.

Mr. Durant spoke on the formation of the International Astronautics Federation and of the work it was doing in the promotion of space flight.

#### **ARS Group Forming in San Francisco**

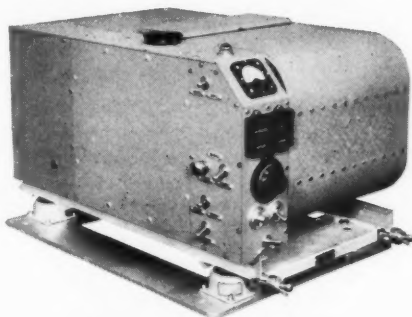
INVITATIONS have been sent to 1800 interested persons to attend the organizational meeting on Nov. 20, 1952, of a new ARS group forming in San Francisco, Calif. A. L. Antonio, of the Aerojet Engineering Corporation, and winner of the 1952 C. N. Hickman Award, will be the guest speaker for the meeting.

A full report of this meeting will be included in the next issue of the JOURNAL.

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# Technical Literature Digest

H. S. SEIFERT, California Institute of Technology, Associate Editor

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## Book Reviews

RADIO ASTRONOMY, by B. Lovell and J. A. Clegg, John Wiley and Sons, Inc., N. Y., 1952, 238 pp. \$4.

Reviewed by J. L. GREENSTEIN  
California Institute of Technology

This interesting book describes the techniques, results, and theories of the new and rapidly growing field of radio astronomy. After pioneer investigations in the United States (Jansky in 1932 and Reber in 1940), the center of activity shifted to England (Cambridge, Manchester) and to Australia. Excellent bibliographies and summarizing reports have been issued (Cornell, Sydney, URSI). Lovell and Clegg have published the first nontechnical survey, and it can be recommended as of very general interest.

The book begins with a brief outline of elementary astronomy and of the methods of active (radar) and passive radio detection of celestial objects. It describes the elaborate antenna arrays, large parabolic collectors, and interferometer arrays now in use; the very recent phase-switching interferometer (Ryle) is omitted. The elegant means are described which permit accurate measurement and position determination on signals which are only one-thousandth of receiver noise power; recent work carries below this level.

The interests of the authors result in a somewhat disproportionate emphasis on the results of radar observations of meteor distances, velocities, and orbits. Daylight meteor showers exist of greater activity than any of the well-known night showers. Meteor velocities have been shown all to be less than the parabolic limit; i.e., meteors belong to the solar system. Interesting theoretical and observational studies of the ion cloud produced by the meteor give new information about the physics of the ionosphere and about high-altitude winds.

The radio noise generated by the sun is discussed with respect to (1) the "quiet" solar thermal radiation from the corona, and (2) the fluctuating sunspot, flare, and coronal noise bursts and storms that reach

enormous intensity (equivalent to thermal emission at  $10^{12}$  or  $10^{14}$  K). Solar radio noise can jam radar and television and must ultimately be taken into account in long-range rocket or satellite control systems.

To this reviewer the astronomically most interesting work is the detection of radio noise in the galaxy and from "point sources." This section is least well covered since it is expanding so rapidly. About 100 small radio sources have been found; many are at great distances. From one tentatively identified at 30 million light years more radio noise is received than from the sun. An important new field is the diffraction of the radio waves from these sources by ionosphere clouds of nonuniform phase retardation. Several "point sources" have recently been identified as peculiar filamentary clouds of interstellar gas or as colliding galaxies. Theoretical explanations involve high-velocity collisions of ionized magnetized gas.

So far only one emission line has been found, that of the hyperfine transition of interstellar hydrogen at 1421 megacycles. New techniques are being developed to permit frequency scanning for other lines and for gas-cloud velocities. A most extraordinary new plan is the construction of a movable 250-ft parabolic collector at Manchester. It is hoped that such new techniques will permit continued rapid progress. Meanwhile, the achievements of the first ten years of a new science should make fascinating reading for a large audience.

THE INITIATION AND GROWTH OF EXPLOSIONS IN LIQUIDS AND SOLIDS, by F. P. Bowden and A. D. Yoffe, Cambridge Univ. Press, N. Y., 1952, 105 pp. \$4.50.

Reviewed by A. J. STOSICK  
Jet Propulsion Laboratory  
California Institute of Technology

This book is a welcome addition to the literature of a field which abounds in rule-of-thumb tests. With considerable justification the book is primarily a collection of the systematic and classical researches of the senior author and his co-workers.

The experiments described are well designed to establish the nature of the important ways of initiating explosions. They have done a great deal toward taking the mysticism out of the behavior of explosives.

The general excellence of the book is disturbed only slightly by an occasional injudicious choice of notation or of argument in some instances.

The argument on page 10 concerning the observed relative effectiveness of electrons, alpha particles, argon ions, and mercury ions based on the size of the particles is not completely convincing. It is quite possible that elementary mechanical considerations of conservation of momentum and energy applied to the collision of the incident particle and the target particle provide a better argument. For an unbound target particle, energy transfer from incident to target particle is greatest when the masses are equal. These considerations have been used with some success in interpreting Szilard-Chalmers processes.

In purchasing a book of this kind, the buyer is not primarily interested in getting a lot of paper for his money, but even in times of inflated prices, \$4.50 seems somewhat expensive for 105 pages of text, references, and index. Apparently the favorable rates of exchange have no effect on the price of Cambridge University Press books sold in this country.

METALS AT HIGH TEMPERATURE, by Frances H. Clark, Reinhold Publishing Corp., N. Y., 1950, 372 pp. \$7.

Reviewed by POL DUWEZ  
Jet Propulsion Laboratory  
California Institute of Technology

This book comprises three sections, dealing with (1) theoretical aspects of creep; (2) testing methods; and (3) properties of metals and alloys at high temperature. The first section is well presented and quite informative, but has little to do with the rest of the book; no attempt has been made to correlate experimental results with basic theoretical principles. The second section is of definite interest to all students and researchers entering the field of creep test-

EDITOR'S NOTE: This collection of references is not intended to be comprehensive, but is rather a selection of the most significant and stimulating papers which have come to the attention of the contributors. The readers will understand that a considerable body of literature is unavailable because of security restrictions. We invite contributions to this department of references which have not come to our attention, as well as comment on how the department may better serve its function of providing leads to the jet propulsion applications of many diverse fields of knowledge.



ing. The third section, by far the most extensive (about three fourths of the book), is a complete compilation of all the mechanical properties of alloys at high temperature. The various types of alloys are dealt with separately in eight chapters. The material is well presented and an accurate list of references will help the reader to find the original articles.

The metallurgist engaged in research on high-temperature alloys will not find much information in this book on the basic metallurgical aspect of such alloys. The wealth of information on properties of alloys at high temperature will be extremely useful to the mechanical engineer responsible for design of high-temperature components. This book will probably become a very popular handbook in all design rooms engaged in jet propulsion research.

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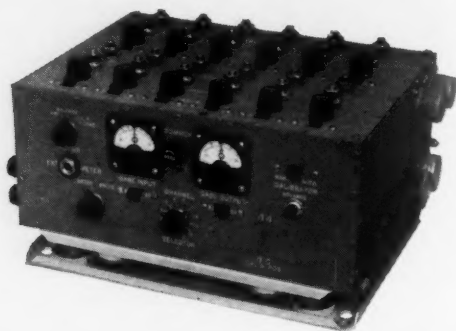
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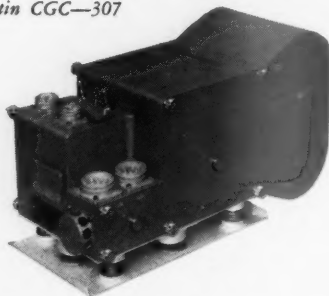
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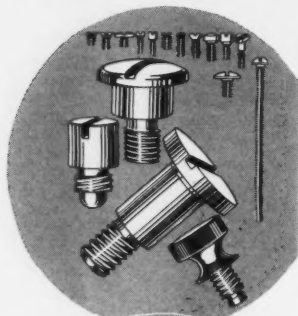
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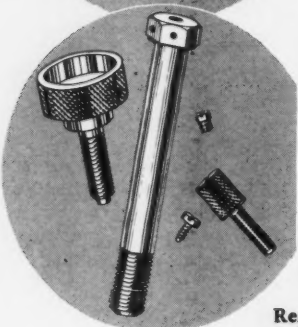
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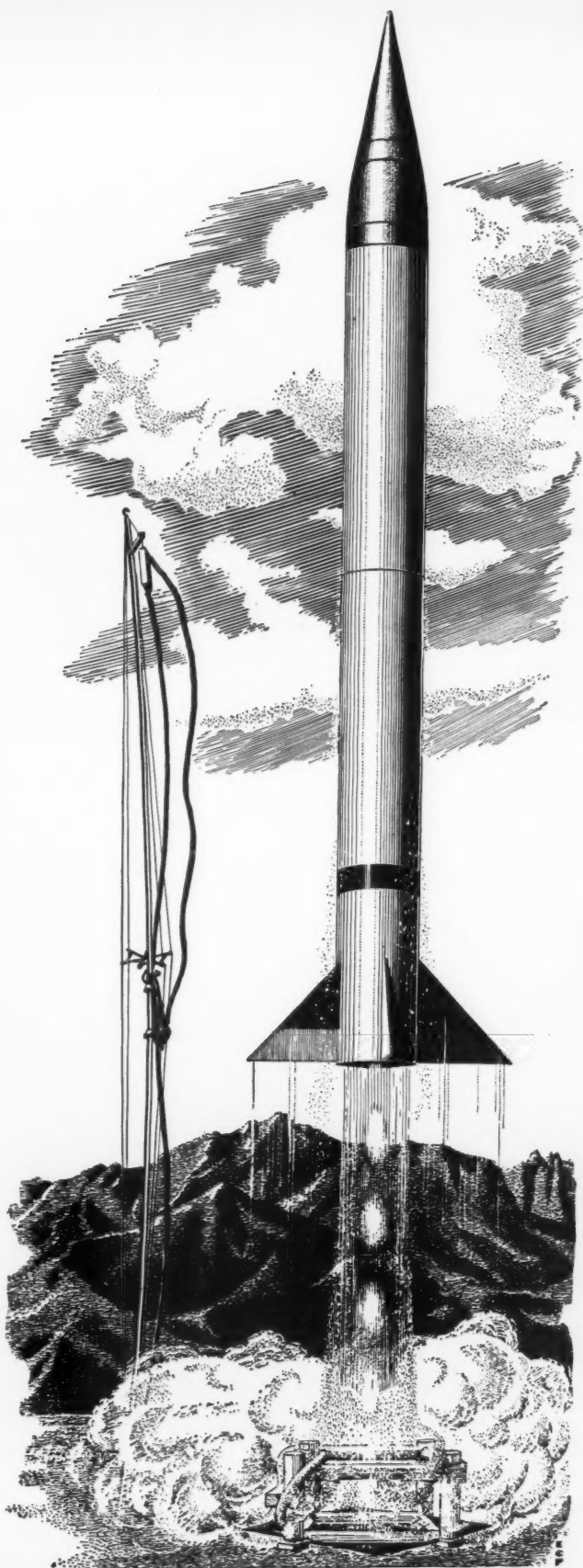


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